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Landing System Study

Final Report

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Autonetics COMPUTERS & DATA SYSTEMS DIVISION
A DIVISION OF NORTH AMERICAN AVIATION, INC., ANAHEIM, CALIFORNIA

Contract NAS2-1201

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Airborne Approach and Landing System Study

Final Report

2 AUGUST 1963 [handwritten initials]

APPROVED BY:

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A DIVISION OF NORTH AMERICAN AVIATION, INC., ANAHEIM, CALIFORNIA

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I. INTRODUCTION

An eventual goal in the operation of piloted aircraft is to provide the capability of instrument flight under zero visibility conditions without the aid of any ground based equipment for all phases of flight. Ideally, this should encompass takeoff from one field to landing rollout at another. This task is not feasible with currently available equipment. As one step toward the solution of the problem and to provide a means for further development, this study was undertaken under the auspices of the Ames Research Center of the National Aeronautics and Space Administration.

The immediate objective is to determine what portions of the task can be accomplished in a practical manner with presently available equipment and to determine the equipment required in the immediate future to implement a system which can be used to investigate and demonstrate various means of accomplishing the desired task. The conditions implied by the statement of the problem have been used as guides rather than limitations, and wherever the intended immediate research application of the system is better accomplished by compromising some of these conditions, compromises have been made.

II. SUMMARY

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A set of ground rules were established to define the scope and framework within which the investigation proceeded. The purpose of the investigation was to consider the feasibility of mechanizing an airborne approach and landing system.

Three generalized system arrangements were considered, differing primarily in the source of aircraft spacial information. One system arrangement uses inertial information, one uses forward looking radar, and one uses onnidirectional Distance Measuring Equipment (DME).

A survey was made of current "off-the-shelf" equipment which might be used to mechanize a system. Component operation and performance were reviewed and tabulated. Criteria were evolved whereby components and systems could be evaluated. Error analyses were conducted which aided in this evaluation.

Recommendations are made concerning the equipment required to implement a system. The recommended system includes a radar altimeter, FM/CW DME, digital computer, two-gyro platform, barometric altimeter and rate of climb indicator, accelerometer, and multiplexer. Specific equipment which meets the requirements was suggested and the source of supply given.

As part of the evaluation, an analog simulation was performed. System operation procedures were finalized. Procedures for manuevers other than those required for approach and landing, as well as flight test procedures were investigated.

The effects of making approximations to the exact equations required to generate the desired display were investigated.

Breadboard circuits were fabricated to confirm the applicability of circuit configurations investigated for generating the display. These circuits consisted primarily of the interface between the computer and the CRT display.

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III. SYSTEM REQUIREMENTS

To establish system requirements and select equipment to implement the system, ground rules were established and criteria were defined to assist in evaluating compliance with the ground rules.

A. GROUND RULES

The ground rules were based upon the original direction of the study or minor redirection which resulted from coordination with Ames Research Center. Criteria were established as a result of past experience, comparison with established current requirements for similar systems, and specific investigations. The ground rules define the scope and framework within which the investigation proceeded and are based upon the following four general premises:

1. The system shall be, as much as practical, airborne.

2. The system shall be capable of driving a display or displays to assist the pilot in making an approach, landing, or go-around under zero visibility conditions.

3. The system shall be made up of equipment which is, as far as possible, currently available and in production.

4. The immediate application and the first implementation will be to use the system as a tool for test and evaluation of displays and techniques in an aircraft of the CV-340 class, outfitted as a test vehicle.

The ground rules are as follows:

1. Wherever possible, equipment should be chosen that is completely airborne. Where this is not possible, the ground based equipment should be small, not require a permanent installation or operator.

2. The system is considered to have a pilot as an integral link in the control loop. Direct coupling through an autopilot will not be considered.

3. The objective of this immediate investigation is to make preliminary definitions of airborne equipment and system configurations which may be flown to investigate, demonstrate, and evaluate

various means of accomplishing the desired tasks, including modes of sensing, computation, and display.

4. The system should be versatile so that landing situation-displays with various formats may be evaluated.

5. Any equipment design or development is beyond the scope of this investigation. Only equipment which is currently available as a production item will be considered, except that interface adapters and couplers, that would normally be specially built items, will be investigated to the extent necessary to provide data for the comparative evaluation of the systems.

6. The pilots display will consist primarily of situation type information. The format of the display will be that developed by Ames. Investigation of the display format will be directed to determine what additional information should be presented, and in what manner this information may be presented to enhance the usefulness of the displayed information.

B. CRITERIA

The system must operate in such a manner that the pilot can easily find the airport, establish a glide path, and maintain vertical velocity and ground track. Further, the system must allow the pilot to aline the aircraft with the runway centerline and execute a satisfactory flare to touchdown. The criteria may be more precisely stated as follows:

1. The pilot must be able to see changes in attitude and flight path of 0.25 deg or less.

2. The angular displacement from the extended runway centerline must be readable to 5 deg or less when the runway lines are out of view.

3. The range from the runway must be directly readable to 1000 ft or less.

4. A difference of 0.5 deg between the aircraft's ground track and the extention of the runway centerline must be discernable by the pilot.

5. The pilot must be able to ascertain a deviation of 0.25 deg from an established glide path.

6. The pilot must be able to note a variation in aircraft glide slope of 0.1 deg.

7. A lateral deviation of 15 ft from the center of the runway must be readily apparent to the pilot when over the runway.

8. The pilot must be presented with information such that he can control the aircraft sink rate to within 2 ft/sec.

9. The pilot must be presented with information such that he can control aircraft heading to within 1 deg of the runway heading at touchdown.

The criteria pertaining to touchdown are most important, and consistently successful touchdowns are dependent upon the continuous availability of accurate information during an approach. Criteria of a more detailed nature are stated and derived in context where they are applied.

IV. SYSTEMS CONSIDERED

A. DESCRIPTION OF THE THREE GENERAL SYSTEM ARRANGEMENTS CONSIDERED

The study was aimed at developing a system to drive a situation display. A pilot should be able to execute an approach and landing using this display. The format of this display was the result of the work of Ames Research Center. This display is similar to that shown in Appendix II in Figure AII-4.

As can be seen in this figure, the runway is depicted diagramatically by the runway lines. The flight path can be inferred from the velocity circle. At low altitudes, altitude is indicated by the altitude dot. A more complete description of the display is contained in Appendix II.

The purpose of this study was to investigate the feasibility of building this type of system, and if it proved feasible, to recommend specific pieces of equipment which might be used to accomplish this task. Several different systems were considered. All these systems include the above mentioned display. This is the heart of the system. The purpose of any other equipment is to drive the display.

In a later section of this report the various systems are compared and a specific one is recommended. This comparison is made on the basis of the performance of various parts of the several systems.

The basic generalized system is shown in Figure IV-1. Three system arrangements were considered. Block diagrams for them are given in Figures IV-2, IV-3, and IV-4. One system uses inertial information, one uses forward looking radar and the third uses DME to determine the spatial position of the aircraft.

Features common to the system arrangements shown in Figures IV-2, IV-3, and IV-4 are:

1. Computer. There is need for flexibility of computation because the system is to be used for evaluation of situation displays useful in landing maneuvers. It is felt that modifications to the initial arrangement will be made during flight test. Airborne analog equipment does not have the required flexibility.

2. Display. This is to be a CRT type of display. Inquiry into the possibility of using electroluminescent displays disclosed that while some form of electroluminescence would probably be more

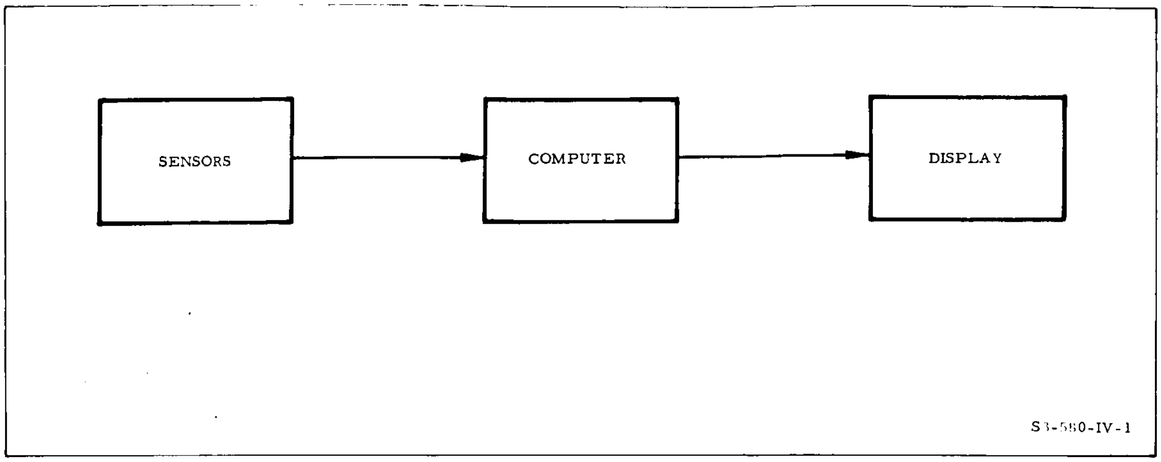

Figure IV-1. Basic Generalized System

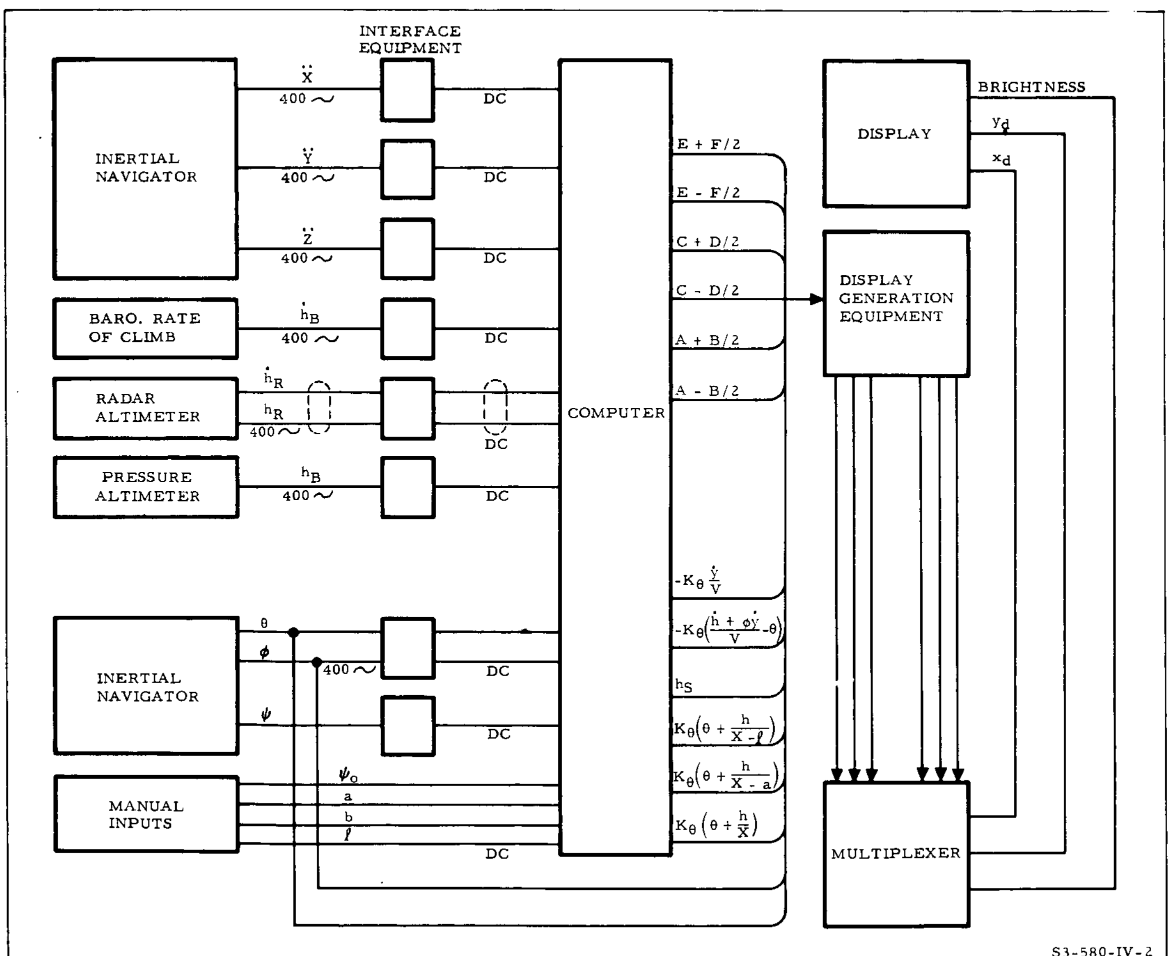

Figure IV-2. System Block Diagram - Inertial Navigator applicable to this type of displayed information at some future time, no electroluminescent equipment is currently available that even remotely meets the requirements. Further, even forms currently in the preliminary development stage are not as applicable as a CRT. Unless some unexpected major advances in the electroluminescent state-of-the-art are made, even prototypes competitive with a CRT would not be available for more than five years.

3. Multiplexer. A multiplexer is used to allow more than one character to be painted on the face of a single beam CRT.

4. Display Generation Equipment. Display generation equipment is used to generate the symbols for display from computed and sensed information. Information for the lines on the display arrives from the computer in d-c form. The information required is the location of the center of the line, the slope of the line, and the width of the line. The signal representing the width of the line modulates a sine wave. The y position of the center is used as a bias. The slope times the width modulates another sine wave and the z position of the center is used to bias this signal. These two signals are used to form a Lissajous pattern which takes the form of a line. Circles are formed by biasing the sine and cosine wave by the y and z positions of the center of the circle. The cosine is formed by passing a sine wave through a phase shift network. Points are formed by directly displaying their y and z positions.

5. Interface Equipment. Interface equipment converts the signal voltages from the sensors to a form acceptable by the computer. It is assumed that the necessary analog-to-digital conversion will be contained within the computer, except as noted below. The function of the interface equipment is to demodulate and properly scale the sensor output voltages.

6. Sensors. The barometric rate of climb sensor, the radar altimeter, and pressure altimeter are common to all of the systems. These are used to derive clean and accurate altitude and altitude rate signals.

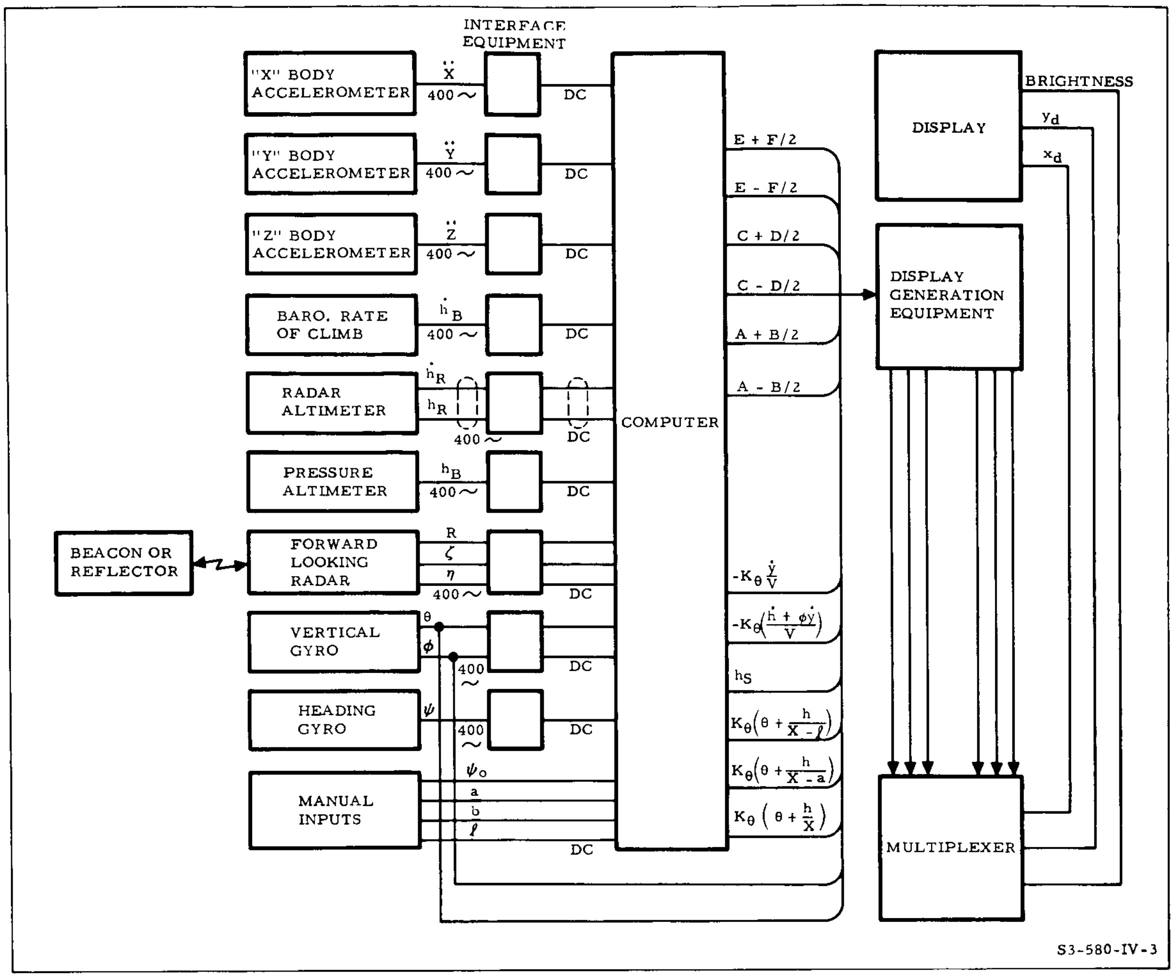

Figure IV-3. System Block Diagram - Forward Looking Radar

7. Manual Inputs. Manual inputs describe the runway, its heading, width, and length.

The primary differences between these three system arrangements are in the sensors used for spacial information. The parameters needed to drive the display are θ, ϕ, ψ, $\dot{h}$, $\dot{y}$, V, X, y, and h. The measurement of θ, ϕ, ψ, $\dot{y}$, and V is normally done with inertial instruments. The parameters $\dot{h}$, X, Y, and h can be measured inertially, by electromagnetic radiation type sensors, or by using a combination of the information from both of these sensors. The parameters h and $\dot{h}$ will be sensed with a radar altimeter at low altitudes.

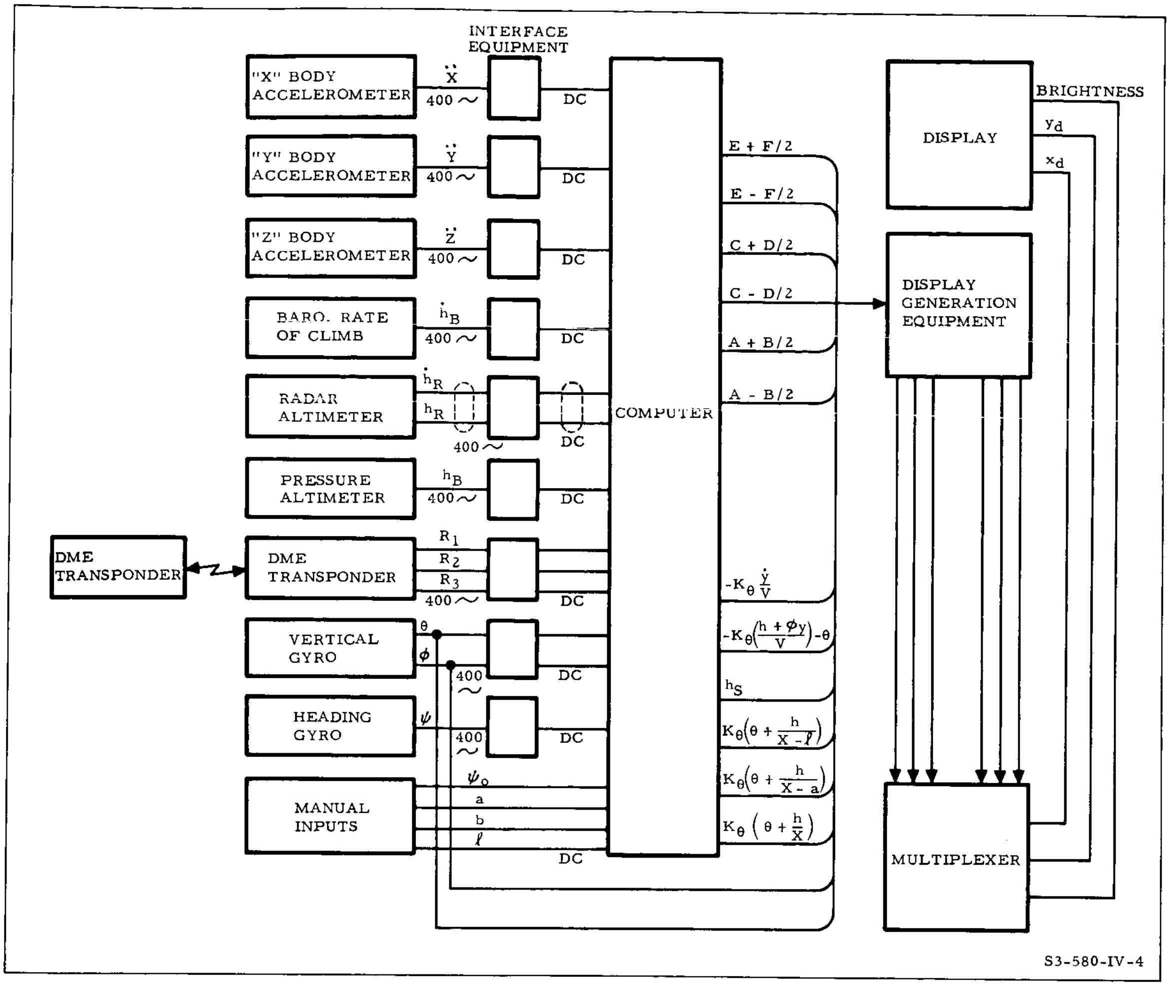

Figure IV-4. System Block Diagram - DME

1. Inertial Platform

The inertial platform is used to supply aircraft attitude and acceleration. The attitude information is usable directly. However, the acceleration information signal must be changed to make it compatible with the computer input. The first integral of acceleration will define velocity and the second integral of acceleration gives displacement. Information from the inertial platform, if sufficiently accurate, is sufficient to be used in the mechanization of the horizon line, velocity circle and runway lines for the display.

2. Forward Looking Radar

The radar set can supply information in the form of range and angle signals which will locate a known point on the ground. Vertical and heading gyros are required to supply attitude information to drive the display.

Three orthoganal accelerometers would be used to provide, after integration, the three velocity components necessary to drive the velocity circle. Information from the attitude gyros would be used to mechanize the horizon line. The combined information from the radar and attitude gyros is required in the computations involved in displaying the runway lines.

3. Distance Measuring Equipment (DME)

DME is used to measure the distance from the aircraft to three or more fixed points on the ground. The position of the aircraft with respect to the runway can be computed from this information. The remaining equipment used with the DME is the same as that used with the forward looking radar.

B. SYSTEM LIMITATIONS

1. All-Inertial Sensor

An accurate assessment of the residual inertial position error, after manual updating of the inertial navigator, has not been made. However, this assessment is dependent upon a manual updating procedure, taking place at relatively short ranges to the runway, that would interfere with performance of the landing maneuver. In addition, the accuracy of new position data would probably be poor unless derived from sources more accurate than the inertial navigator.

The all-inertial system has the drawback that linear distance accuracies are at best marginal for this application. These inaccuracies are primarily due to inaccurate distance information preset into the autonavigator, and to instrument drift rates. If, during the evaluation of the display, the landings are performed at the same airport from which the flight originated, then there would be no errors introduced from the cartographical process. A nominal figure for accelerometer null inaccuracy is 10^{-5}g. Since $d = 1/2\ at^2$, a distance uncertainty of 20 ft could occur approximately 6 min after alinement.

2. Monopulse Radar Sensor

Existing ground tracking radars have not exhibited a high degree of accuracy. The ground tracking process requires maintaining angle and distance information of one target where, for an ordinary airport, many extraneous targets exist. In the presence of this clutter, acquisition and track may be difficult. In some cases, a strong return from a radiation pattern side lobe may cause loss of lock-on. It is clear that a matched beacon target would be desirable. Even when operating satisfactorily, the system will produce significant error in parameters. This is indicated in Appendix III, Error Analysis.

3. DME Sensor

Although the DME configuration provides range information with only small errors, it has a drawback in that three portable ground based transponders are required. These must be placed at known points near the runway.

C. SYSTEM COMPARISON

The several system types were evaluated with respect to accuracies, availability, basic cost, engineering development requirements, weight, physical adaptation to airplane, adaptation to associated equipment, operational ease, reliability, and other limitations. These evaluations are discussed in Sections V and VI and, that the DME sensor configuration would best fit the needs of this mission. Table IV-1 generally summarizes in a very qualitative way, some relative characteristics of the three systems. Each system is given a relative rating in each category of 1, 2, or 3. If a unity weighting factor were applied to each category, then the totals would represent the applicability of each system (low number most applicable).

Table IV-1. System Comparison

	Relative Rank of Configuration								
	DME			Radar			Inertial		
	1	2	3	1	2	3	1	2	3
Accuracy	X				X			X	
Availability	X			X			X		
Basic cost	X			X				X	
Development cost	X				X				X
Upkeep cost	X				X		X		
Airborne weight	X				X		X		
Compatability with aircraft		X				X	X		
Compatability with associated equipment		X		X			X		
Operational ease	X					X		X	
Reliability	X					X	X		
Range limitation	X					X	X		
Totals		13			23			16	

V. COMPONENT SUMMARY

A survey of current "off-the-shelf" equipment which might be used to mechanize an airborne approach and landing system was accomplished. This survey was divided into areas corresponding to the different system functions (see Figure IV-1). Thus areas covered in the survey include sensors, computers, and interface equipment. The results of this survey are given below.

A. MAJOR SENSORS

The major sensors are those that can be used to measure linear and angular position in space. Radar, inertial navigators, gyros, and DME were surveyed.

1. Radar

A radar capable of tracking a ground target would be necessary. When a radar tracks a target, it is continuously following the target both in range and in angle. Most tracking radars are designed to track isolated targets. An isolated target is one which has free space as a background. This might make it necessary to install a beacon on the ground so that the target is distinguishable from the background.

This type of radar must be locked on to the target to obtain electrical signals related to range and angle. This is characteristically done by initially having the radar scan a segment of space. If the target is within this segment, the operator positions two cursors over the target and initiates tracking with a switch. If the target is not within this first segment of space, the operator must position the antenna to a new position in space and resume scanning.

Once the radar is locked on, operation is automatic. The outputs of the radar are generally slant range, azimuth angle, and elevation angle. Tracking may be discontinued if the radar loses lockon. This may occur if the radar was not locked on accurately initially, if the dynamics of the aircraft maneuvers are excessive, or if system noise is of a sufficient magnitude to make the target indistinguishable. This latter case can arise from atmospheric effects, the characteristics of the background, or if an active beacon is used, by echoes. Once tracking is discontinued, lock on must be manually accomplished again as outlined above.

Information is available (most of which is classified) on airborne tracking radars manufactured by Hughes Aircraft Company (MG-12, MG-13, MA-1, AN/APG-51C, AN/ASG-18 armament control radars), Westinghouse (AN/APQ-50 and AN/APQ-72 armament control radars), Magnavox (AN/APQ83 armament control radar), North American Aviation (R-14 and F-15 radars), RCA (AN/ASG-14 and AN/APQ-35 armament control radars), and Stewart-Warner (AN/APG-53 radar). Typically, these radars have the following characteristics:

Weight 100 to 1000 lb

Frequency X Band

Peak power 0.2 to 250 kw

Pulse repetition frequency 300 to 250,000 pps

Antenna size 1 to 3 ft maximum dimension

Beam width 1 to 7 deg

Maximum tracking range 2 to 50 naut mi

Maximum track rate 100 to 2100 knots

Other typical characteristics of interest are that accuracy is nominally ±0.1 percent of range and angle accuracies are of the order ±1/2 deg. Also, the antenna is physically constrained so that it can only cover, at most, ±90 deg in any direction.

2. Inertial Navigators

The heart of an inertial navigator is the stable platform. In one form, the navigator maintains the platform perpendicular to the local vertical. Three gyros are mounted on the platform. These sense heading or yaw angle, roll angle, and pitch angle. On some navigators, the platform is forced to maintain a constant heading. There are provisions for mounting accelerometers on the platform. The outputs of these accelerometers can be doubly integrated to give aircraft displacements. Where the platform is forced to maintain a constant heading, these displacements are in normal inertial coordinates.

Normally the navigator is alined just prior to usage. This process can be quite lengthy. As soon as the alinement process is completed, errors start propagating. In the gyros, this is due to drift and can be quite small, of the order of 10^{-3} deg per hour. The double integration of acceleration to give distance also is prone to errors. A constant combined error equivalent to 2×10^{-6}g would give rise to a displacement error of

10 ft at the end of 10 min. (This error is the combination of all accelerometer errors and interface and integrator errors reflected back to the input of the integrators.) Inertial systems with this low an error do not exist.

Minneapolis-Honeywell and North American Aviation were contacted with regard to using their inertial navigators for this application. Assurances were received that although their equipment was of the finest quality, it would not be satisfactory because of this linear measurement problem. This opinion was voiced even when one of the ground rules was relaxed to the point of requiring the system to operate only for the go-around period between takeoff and landing at one airport under test conditions.

3. Gyros

In the event that an inertial navigator is not used other means of sensing vehicle attitude and heading must be used. There are several airborne methods of obtaining a directional reference. These are:

1. Magnetic compass
2. Directional gyro
3. Directional gyro systems with magnetic compass slaving
4. Two-axis gyro flight reference platforms
5. Gyro compass systems

a. Magnetic Compass

The magnetic compass or remote-reading flux valve is subject to errors of a nature outside the consideration of this report. Geographical magnetic variations, local deviations due to electrical fields and metal parts of the vehicle, and northerly turning errors produce uncontrollable long and short-term errors in navigation.

b. Directional Gyro And System

The directional gyro, in its more accurate configuration, provides one of the best sources of directional information, especially for flights of short duration. Accurate initial orientation is required before the gyro is uncaged for the flight. Accuracy of heading during the flight is then a function of gyro random drift for short flights plus earth's rotation, geographical latitude, and the coriolis effect. Electrical gimbal torquing motors coupled with a gravity sensor are generally used to initially orient the spin axis, inner and outer gimbal axes mutually orthogonal. The outer gimbal with its 360 deg of angular displacement is used to indicate direction or azimuth through the output of a pickoff while the spin axis remains fixed in inertial space.

The latest directional gyro configurations generally employ a means of rotating or oscillating the gimbal bearings and gimbal wiper assembly to virtually eliminate static friction torques and thereby reduce random drift to a value below 0. 05 deg/hr in some cases.

Compensation for earth's rotation is usally provided by a manual setting of geographical latitude on a control panel. This also provides a meridian convergence correction dependent upon the latitude set in.

The conventional ball bearings on the spin motor may be replaced with gas bearings to ensure longer life and improved performance.

The directional gyro may be slaved to a magnetic flux-valve if so desired in connection with longer flights. The flux-valve is in effect a magnetic compass device with an electrical pickoff which transmits heading information from a remote location in the vehicle. Compass deviation effects are thereby decreased due to elimination of compass proximity to radio fields and cockpit disturbances. When used in this manner, the random drift of the directional gyro is eliminated as a time function; but long range compass errors exist due to inequalities of magnetic variation along the course, for which it is impossible to provide continuous adequate correction biases. The use of the slaved gyro, however, is an improvement over the compass alone, as the gyro stabilizes the course reading and eliminates compass oscillation effects, particularly during banks and turns.

c. Two-Gyro Reference Platform Systems

Random drift and overall directional accuracy of a directional gyro system may be improved by mounting the two-axis directional gyro on a gimbaled platform together with a two-axis vertical gyro. By maintaining the vertical accuracy of the platform, the performance of the directional gyro is greatly increased.

System inputs and flux-valve are used as described in Par. b. above.

d. Gyro Compass-North Seeking Gyro

The true gyro compass as used for many years in marine navigation is neither a directional gyro nor a magnetically slaved gyro. It is highly accurate, being able to sense true north to within a small fraction of a degree. There is no variation error due to the difference between magnetic north and true north nor is there any deviation due to magnetic fields or metal parts in the ship. Use in aircraft has been lagging due to the lack of accurate ground speed information.

Some other advantages over the directional gyro or slaved gyro are listed below:

1. Not affected by magnetic storms.
2. Good up to ±80 deg latitude.
3. May be used as a free gyro in polar regions above ±80 deg latitude.
4. Continues to work during turns.
5. Need not be set to accurate north heading at start of flight, as is required by the directional gyro.
6. No cumulative drift. Average heading is always true north and net accuracy increases with flight time.
7. Latitude correction is not critical as in the directional gyro.
8. Not affected by gyro mass shifts which cause drift errors in the directional gyro.

The gyro tends to precess and hold its spin axis in the plane of the earth's spin axis. This is effected by its coupling with earth's rotation. Assuming the spin axis to be pointing north and parallel to the earth at a latitude angle λ , and the tilt axis pointing east and west and parallel to the earth, then earth's rotation will appear to make the spin axis rotate in azimuth (eastward in northern latitudes) at an approximate rate of $W_E \sin \lambda$ where W_E is earth's rate of rotation in radians per second. At the same time, the spin axis will start to tilt (eastward end up, west end down) at the approximate rate of $\phi\ W_E \cos \lambda$ where ϕ is the angle of azimuth the spin axis has rotated. In order to keep the spin axis pointed north a voltage proportional to the rate of tilt is fed into a torquer motor on the tilt axis to produce a torque T_H opposing the gyro torque $H\ W_E \sin \lambda$ caused by earth's rotation, where H is the gyro angular momentum. This torquing force will act upon the vertical axis and precess the spin axis back to true north where it will continue to point as long as earth's rotational effect is balanced by the tilt angle torquing force. This correction also is applicable to directional gyro systems except that small latitude errors produce drift problems.

We now have a gyro compass that will point north while located at a specific fixed point on the earth. If the gyro is mounted in a fast moving vehicle, such as an aircraft, an accurate north and east component of velocity will be required to provide compensation.

North velocity error, caused by the apparent slowing of the eastward motion of the earth as the vehicle passes through higher latitudes can be compensated by a bias torque about the vertical axis of the gyro. To the north moving gyro compass, the earth appears to be rotating about an axis displaced from its real axis at an angle e, and a westward error in the compass is produced. This angular error e can be shown to be:

$$e = \tan^{-1} \frac{v \cos C}{R\, W_E \cos \lambda + v \sin C} \qquad (1)$$

where v = aircraft velocity (ground speed)

C = aircraft course

R = radius of earth (20.9×10^6 ft)

W_E = earth's sidereal rotational rate

λ = latitude of vehicle, and

$R\, W_E \cos \lambda$ = eastward velocity of earth's surface at latitude λ.

A correction torque $T_v = \dfrac{H\, v \cos C}{R}$ is therefore (2)

applied to the gyro's vertical axis, where $v \cos C$ is the north component of the vehicle velocity and H = angular momentum of the gyro wheel. It is noted that the angular error e goes to zero on east and west courses, as cos 90 or cos 270 is zero in the numerator of equation (1).

East velocity error is produced by the east or west velocity of the vehicle causing an apparent increase or decrease of the earth's eastward angular rotation, W_E. This velocity change is equal to $\dfrac{v \sin C}{R \cos \lambda}$.
Because the compass vertical axis component of the earth's rotation depends upon latitude λ, the above equation multiplied by $\sin \lambda$, or

$$\frac{v \sin C \sin \lambda}{R \cos} = \frac{v \sin C \tan \lambda}{R} \qquad (3)$$

defines the amount of the vertical component error. The amount of net horizontal torque T_H to be applied must then oppose the error of equation (3) added algebraically to the earth's rotation $W_E \sin \lambda$. Multiplying by H we then have the net torque:

$$T_H = H W_E \sin \lambda + \frac{H v \sin C \tan \lambda}{R} \tag{4}$$

In slow moving vehicles where the velocity is small and high accuracy is not required, the second term may be considered zero. In aircraft, however, the second term must be used or a large navigation error will result.

Other errors occurring in the gyro compass are: (1) ballistic deflection error caused by accelerations during changes in vehicle speed and direction, (2) ballistic damping error, and (3) centrifugal and Coriolis errors. Corrections required are dependent upon the speed range of the vehicle in which the compass is used. Slow speed aircraft would probably require corrections for (1) and (2) above, plus the Coriolis error. It should be noted that the aircraft velocity is available since it is a parameter necessary for the display.

e. Vertical Gyro

If the magnetic compass, directional gyro, or gyro compass are used, some means of sensing aircraft pitch attitude is necessary. This is usually done with a vertical gyro. A vertical gyro is a two-degree-of-freedom instrument whose gimbal displacement about each output axis constitutes a measure of angular deviation from the local vertical axis. The rotor spin-axis is maintained parallel to local vertical by a gravity sensing device. The term vertical gyro derives from the fact that it is used to measure displacement from the vertical reference. A vertical gyro effectively serves the same purpose as a pendulum, with the advantage that a maneuver does not cause it to oscillate. This advantage is particularly desirable since attitude information is most needed during maneuvers.

A high erection rate results in good repeatability. However, it makes the gyro more susceptible to those dynamic errors which accompany aircraft accelerations. The selection of an erection rate in any particular application is a compromise between repeatability and dynamic error.

There are many manufacturers of these types of equipment. The following list is meant to be representative of the equipment available. (Prices given are approximate and are intended to reflect relative price range only.)

4. Heading Systems

a. Sperry Phoenix Company, Division of Sperry Rand Corp

(1) C-12 Gyrosyn Compass Systems. A description of the C-12 gyrosyn compass system follows:

1. Price $7000 to $8000. C-12 Directional Gyro only, $3500.

2. Operation. The C-12 is Sperry's latest and most accurate directional compass system. The basic unit, the directional gyro is built with "Rotorace" or rotating bearing races on the gimbal axes which reduce static friction torques and produce very low random drift rates of less than 1/4 deg per hour, rms.

   A digital controller panel presents a digital readout of heading to 0.1 deg, and provides latitude control inputs, hemisphere selector, magnetic slave control, etc. A compensated flux-valve, included in the system, provides magnetic heading for slaved gyro operation. Coriolis compensation is furnished to the flux-valve by a remote compensation package. Compensation mechanism for drift, earth's rate, meridian convergence, and roll gimbal error is included in the gyro. Sperry advises that a gyro compass mode of operation is under development in connection with this system.

3. Performance. Magnetic slaved accuracy 0.25 deg rms

   Free gyro operation:

Random drift:	0.25 deg/hr rms
Weight:	21.7 lb
Power requirement:	115 v, 400 cps single phase, 65 va running 80 va starting

4. Outputs. 5 heading transmitters
Independent gyro output

5. C-12 System Component Part Numbers.

C-12 Directional Gyro	2586333-1
Flux-Valve	1775277

Remote Magnetic Compensator	2586257-1
Digital Readout Controller	2586261-1
Amplifier and Power Supply	2586384-1

(2) Rotorace Inertial Directional Gyro (RIDG) Platform Systems - SYP-800 and SYP-850. A description of the Sperry RIDG platform systems follows:

1. Price. Up to $22,000.

2. Operation. The RIDG Platform systems by Sperry have been produced to provide heading accuracy commensurate with doppler velocity accuracy for navigation. The Rotorace gimbal bearings provide performance close to that of a floated gyro inertial platform. The vertical gyro provides an accurately stabilized platform to support the directional gyro and thereby improve its free drift performance. Otherwise the system configuration and ancillary components are similar to the C-12. Correction inputs and self-test capabilities are optional.

3. Performance.

(1)	SYG 620 directional gyro random drift:	0.05 deg/hr
	Mass unbalance:	0.1 deg/hr/g
	Anisoelastic drift:	0.05 deg/hr/g^2
	Repeatability:	0.03 deg/hr
	Torquer linearity	0.02 percent

		Roll	Pitch
(2)	Vertical gyro random drift:	2 deg/hr	1 deg/hr
	Mass unbalance:	3 deg/hr/g	2 deg/hr/g
	Anisoelastic drift:	2 deg/hr/g^2	1 deg/hr/g^2
	Torquer linearity:	1 percent	1 percent

(3) System Weight. (Platform, Controller and electronics) 32 lb

b. Lear-Seigler, Inc.

(1) Platform System 2171AB. Used in Compass Systems AN-ASN-37 and LS-300. A description of the Lear-Seigler all attitude two-gyro master reference platform system follows:

1. Price. $15,000 with electronics

2. Operation. This platform provides all attitude capability by the incorporation of a redundant outer roll gimbal. Operation as a directional system is similar to that described above.

3. Performance.

(1) Directional gyro - rotating case construction
Magnetic slaving accuracy: less than 0.75 deg rms
Random free gyro drift: less than 0.25 deg/hr rms

(2) Vertical gyro - rotating case construction
Free drift: less than 4 deg/hr rms
Verticality maintained to: ± 1 deg

(3) The Lear rotating case construction of the gyro gimbals creates a constant rotation of both the bearing races and the slip-ring wipers, thereby reducing static friction to zero and lowering random drift by an order of magnitude.

NOTE: A Lear publication, "LS-300 Reference and Navigation for C141 Aircraft" is available from Lear-Seigler.

4. Component Part Numbers

Platform:	2171-AB	
Power supply and amplifier:	3311-H	
Compass adapter:	6501-T	
Controller:	3804-C	
Lear rate switching gyro:	7650	$350
Flux-Valve - any available commercially		$350
Indicator - any available commercially		$600

The above parts approximate the AN/ANA-37 as used by U.S. Air Force and U.S. Navy.

(2) Directional Gyro. Lear is also able to furnish their STEEL GYRO in directional configuration with rotating case concept with a random drift of 0.05 deg/hr rms.

c. Kearfott Division, General Precision, Inc.

(1) Directional Gyro. The Kearfott Directional Gyro #C70-2205-001 incorporates oscillating gimbal bearings and will meet a random drift rate of 0.25 deg/hr rms. The price is approximately $2000.

(2) Heading and Vertical Reference Platform. The price is approximately $35,000. Other characteristics follow:

1. Operation. This platform incorporates an oscillating bearing, non-floated attitude two-axis directional gyro plus two floated rate integrating gyros to sense roll and pitch. However, cost is high.

2. Performance

Azimuth:	0.25 deg/hr rms drift
Pitch:	0.50 deg/hr rms drift
Roll:	0.50 deg/hr rms drift

This platform can be used with other components to provide a complete directional system. The Kearfott Division will tailor their systems to meet customer requirements.

d. The Bendix Corporation, Eclipse Pioneer Division

(1) Two-Gyro All Attitude Control - Contemplated. The present Bendix platform does not meet the drift requirements of less than 0.25 deg/hr in azimuth. However, Bendix advises that they have an improved two-gyro platform under development incorporating rotating gimbal bearings on the non-floated vertical and directional gyros. The present drift rate in azimuth is 1 deg/hr.

e. American Gyro, Division of Tamar Corp.

(1) Directional Gyro - Contemplated. American Gyro does not, at present, have a directional-gyro configuration of their two-axis FB45-SM2 floated attitude gyro as used in the GAM-77 missile. They advise, however, that they are contemplating a modification program whereby oscillating gimbal bearings and slip-ring wipers will be employed to reduce present drift to 0.1 to 0.25 deg/hr, 100 percent floatation, temperature stabilization and air spin motor bearings will be employed and spin motor speed will be increased from 24,000 rpm to 48,000 rpm in order to obtain a higher angular momentum without weight increase. The price would be approximately $6000. This unit will be especially ruggen and due to its floatation, will be able to withstand high shocks to 100 g's and vibration over 10 g's to 2000 cps.

f. Autonetics, Inertial Navigation

(1) Directional Gyro - Contemplated. A modification of the present MINUTEMAN air supported free rotor gyro is being contemplated in order to produce relatively low priced vertical and directional gyros. Directional accuracies in the order of 0.1 to 0.3 deg/hr drift are expected. The gyro should sell at approximately $1800 in production quantities.

g. American Bosch Arma Corp.

(1) Arma Subminiature Gyro Compass. A discription of the Arma subminiature gyro compass follows:

1. Price. $14,000 in small quantities; $8000 in production.

2. Operation. Arma has built over 50 units of their true-north seeking gyro compass under contract. Improvements are under way to reduce original warmup time and size. This unit is built with the correction inputs described above as required for air vehicle use. In particular, speed inputs of up to 200 knots will be acceptable to provide north and east vehicle velocity components. Heaters are used to ensure temperature stability and fast warmup. Vibration isolation mount is available for severe aircraft environments. If the compass is used above ± 80 deg of latitude, the gyro should be switched to a directional gyro mode.

   The two-degree-of-freedom gyro unit is supported in fluid at neutral buoyancy and centered by means of fine wire filaments. Friction torques are therefore entirely eliminated as would not be the case if bearings were used. Earth's gravity is determined by a highly sensitive pendulum.

   No periodic maintenance is required as the gyro motor is hermetically sealed and electronic components and transistor amplifiers are all conservatively rated.

3. Performance and Characteristics.

   New size reduced to 9 in. diameter x 11 in. high

Readout:	Direct dial plus synchros
Operating temperature:	-65 to 130 F
Latitude range-compass:	± 80 deg Latitude
Directional gyro unlimited	
Speed inputs:	Modified for up to 200 knots
Power warmup:	225 w max.
Power operating:	30 w
Latitude correction input:	Manual
Warmup time:	Reduced to 10 to 15 min
Turntable error:	±0.5 deg average
Small angle scorsby error:	±0.5 deg average
Large angle scorsby error:	±1.0 deg average
Directional gyro error:	±0.25 deg/hr drift

5. Vertical Gyros

a. American Gyro, Division of Tamar Electronics, Inc.

Model number	VG 45
Maximum weight	10 lb
Angular freedom roll	360 deg
Angular freedom pitch	±82 deg
Repeatability to established vertical (1/2 cone angle)	6 min
Free drift rate	0.12 deg/min
Erection rate	5 deg/min (fast), 1.3 ± .5 deg/min (slow)
Motor input voltage	115 v
Frequency	400 cps
Number of phases	3
Transmitter type	synchro
Nominal excitation	26 v
Frequency	400 cps
Voltage gradient	206 mv/deg
Output characteristics	sine wave

b. Kearfott Division, General Precision, Inc.

	Model Number		
	T2104-2A	423418	T4100-3C
Max. weight (lb)	8	7	9
Angular freedom (roll)(°)	±85	360	360
Angular freedom (pitch)(°)	±45	±85	±85
Vertical reference	Electromagnetic	Electrolytic	Electrolytic
Repeatability to established vertical (min. 1/2 cone angle	1.5	15	15
Free drift rate (°/5 min)	1.5	2	2.1
Erection rate (°/min)	1.5 ± 0.5	2 ± 1	$1^{+0.5}_{-0.25}$
Synchro null (mv max.)	15	----	35
Motor input voltage (volts)		115	
Frequency (cps)		400	
Number of phases		3	
Transmitter type	2-wire synchro	Potentiometer	2-wire synchro
Nominal excitation (volts)	26	5 v dc	35
Frequency (cps	400	---	400
Voltage gradient (mv)	3600	.5 & 0.028 roll 0.061 pitch	105
Output characteristics	sawtooth	---	linear to ±60 deg
Output voltage (volts line to line)	45	5 v dc	11.5

6. DME

There are two types of DME. One is a pulsed type of device which operates in a way similar to radar. Its accuracy is not as good as radar. The other type of DME uses FM/CW techniques. This type of device is made with accuracies which are phenomenal when compared to the maximum range of the equipment. It is available off-the-shelf, and is reasonable in price.

Two such devices were investigated. One is manufactured by a division of ITT; the other by the Cubic Corporation.

The CW DME measures the phase shift which takes place in a two way transmission and calculates range from this phase shift. This is done by modulating the carrier with several tones. The tones are selected such that the wavelength of the longest is such that no ambiguities in the range measurement can occur, while the shortest wavelength is such that the accuracy requirements can be met. The total signal is transmitted to a transponder on the ground whose location is known. The transponder retransmits this signal, after a fixed interval, to the airborne transceiver. The signal received at the airborne station is demodulated and the phase shift of each of the component tones is measured. This is then converted to range which is the output of this sensor.

The device manufactured by ITT is such that for each ground based transponder, there must be one airborne transceiver. Since three separate measurements are necessary to locate the aircraft, three transponders and thus three transceivers would be necessary (excluding the possibility of using an independent altitude measurement).

Cubic has several FM/CW DME systems in production. They all are capable of interrogating three transponders with a single airborne transceiver. It also appears that the equipment of one system is compatible with that of another. In this way, a great many special requirements can be satisfied with off-the-shelf components.

The following list comprises the applicable CW DME equipment surveyed.

1. Cubic Corporation

The AERIS System performance is as follows:

a. Distance measured 10 meters to 250 km
b. Accuracy 1 meter

c. Number of simultaneous measurements - four
d. Communications provisions voice channels from all responders to interrogator except while measuring range
e. Operating frequencies 260 and 300 mc

2. ITT

The UHF-200D performance is as follows:

a. Distance measured 25 meters to 25 km
b. Accuracy 2.5 meters or ±0.05 percent, whichever is greater
c. Number of simultaneous measurements one
d. Communications provisions none
e. Operating frequencies 400 to 500 mc

B. SUPPLEMENTARY SENSORS

The supplementary sensors are a radar altimeter and accelerometers.

1. Radar Altimeter

Some of the altimeters investigated are special purpose types which were developed for specific requirements. Included in the group are three units for use with space probes, two for hydrofoil devices, and one for an altitude rate sensor. For the most part, these are engineering models and prototypes.

Those manufacturers in production, or limited production, include the following:

Emertron	(1180, 1280, 1380)
Sperry	(AN/APN 150)
Honeywell	(7091, 7182)
Giannini	(All models limited assembly)

The Emertron ERP-3931-31 low altitude sensing device for GAM-77 was not included in this study but is currently in production.

Cost per unit (small quantity) varied from $5800 for the Spacors 62-A rate sensor to $8000 - $10,000 for the Intec and Sanders altimeters to $19,000 for Emertron 1480. Sperry feels that once full production is reached and development costs are prorated that an APN-150 may cost approximately $9500, including indicator.

Table V-1 lists the radar altimeters surveyed.

Table V-1. Radar Altimeters

Giannini	WRA711X	FM	4300	Binary	20 mw	1-500 ft	±0.1% or 1 ft	0.5-25K ±1%	Self Cal	3 lb	30°ε	30 w	Horn
Giannini	WRA712X	FM	4300	Binary	20 mw	1-5000 ft	±0.1% or 1 ft	0.5-25K ±1%	Self Cal	4 lb	4.5°	30 mw	20" parabola
Sanders	XP(F100)	FM/CW	1630		750 mw	0-5000 ft	±3% ind.		Fail Relay	8 lb	50°	35 w	Flush Mt. Slot
(Argos Prop) Emertron		FM/CW	4300		250 mw	5-1500 ft	±2 ft + 2% ±2 ft to 40'	±150'/sec ±1 ft + 2%	Fail Relay	9.9 lb	60° Pitch 90° Roll	60 w	2 Flush Mt.
(Proposed) Sperry	AN/APN-150	FM/CW	4300			0-5000 ft	±5% to 5K'	±150'/sec	Fail Relay	17 lb	25° Pitch 30° Roll	100 w	
Honeywell	7091	Pulse	4300		100 w	0-200 ft 0-2000 ft	1 ft ± 1% 10 ft ± 1%	50'/sec 1 ft ±5% 500'/sec 10 ft ±5%	Fail Relay	16 lb.	20° Pitch 45° Roll	100 w	Dual Horn
Honeywell	7182	Pulse	4300		50 w	0-500 ft	15 ft ± 0.01 h	50'/sec 10% ± 1.5 ft	Fail Relay	19 lb.	50° Pitch 50° Roll		Dual Plate
Honeywell	YG784					0-5000 ft							
Emertron	APN 42	Pulse	4225		1 kw	500-60,000	±(25' + 0.025%)		Fail Relay	37.1 lb.	17°H 24°E	300 w ac 22 w dc	Dual Array
Emertron	APN120	Pulse High FM/CW low				500-75,000 0-3000 ft	22 ft ± 0.02% 2 ft ± 2.5%		Fail Relay	63 lb.	17 P&R 34°P-65°R	350 w ac 42 w dc	4 Antenna
Emertron	1180	FM/CW	4300	h&h volts 7 switch pts	300 mw	0-3000 ft	2 ft or 5%	±150'/sec 5%	Fail Relay	23.75 lb.	25°P 25°R	56 w ac 60 w dc	Horn Waveguide
Emertron	1280	FM/CW	4300		325 mw	0-5000 ft	2 ft or 5%	±150'/sec 5%	Fail Relay	19.9 lb.		35 w ac 75 w dc	Waveguide
Emertron	1380	FM/CW	4300		300 mw	0-5000 ft	2 ft or 5%	±150'/seo 5%	Fail Relay	19.6 lb.		35 w ac 75 w dc	Horn Waveguide
Intec	AM220H	FM/CW	1630		1 w	0-950 ft	±3' 0-100 ±3% 100-950		Fail Relay	20 lb.	±40°P ±40°R	2 a 28 v	2 Dual Array streamline plates
R. C. A.		FM/CW	1630		750 mw	0-5000 ft	±2' or 3%		In Flight & pre-flight self-test	10 lb. W/IND	±30°P ±45°R	35 w ac	Spiral circuit Flush Mt. Horn
Bendix	AN/APN-141	Pulse	4300		1 kw	0-5000 ft	±2' or 5%		Push-Button self-test	115 lb. W/IND	±30°P ±40°R	70 w 1 ph	Flush Mt.
Bendix	AN/APN-146	Pulse	4300		1 kw	0-5000 ft	±2' or 5%		Push-Button self-test	16 lb. W/IND	±30°P ±40°R	75 w 3 ph	Flush Mt.
I. T. &T.	RA 40	FM/CW	4300		500 mw	0-500 ft	±3' or 3% 1' at 0		In Flight self-test	27 lb.	±17°P ±20°R	60 w ac 120 w s.t	2 Horn
Sanders	TA-101 TA-104	FM/CW	1630		1 w	0-3000 ft	±1' 0-20' 5% 20'-3K'	Optional extra - available	Fail Relay	6.2 lb.		28 w	Flush Mt.
Sylvania	APN-138/139	FM/CW	4300			0-1000 ft				18.5 lb			
Spasors	62-A	CW	X, X, Ka					±30'/sec ±0.2ft/sec	Rate Only	25 lb.	±30°P ±30°R	0.9a ac	Dual Horn Waveguide
Emertron	1480	FM/CW				10-50 ft	2%		Hydrofoil Opr.			20 w ac 2 a dc	Dual Horn
Ryan		AM/CW	13.3 kmc			5-25 ft	±3 in.		Hydrofoil Opr.	15 lb		5 w dc	Dual Horn
Giannini	WRA611X	Pulse	9300		500 w	5K-200K ft	±100'		Self Cal	6 lb.	45° Cone	50 w	20" Parabola
Giannini	WRA612X	Pulse	9300		4 kw	5K-400K ft	±100'		Self Cal	7.5 lb.	45° Cone	70 w	20" Parabola
Giannini	WRA611K	Pulse	16 kmc		5 kw	5K-400K ft	±100'		Self Cal	3 lb	2.5° Cone	80 w	

*Linear outputs unless otherwise noted.

2. Accelerometers

Accelerometers will be used as sensors to obtain velocity information. In the case where an inertial navigator is used, the accelerometers which are part of this device would be used as the primary sensors from which to obtain velocity. If radar or DME is used, accelerometers would be used to improve the short term accuracy of the velocity measurement.

Force balance accelerometers consist essentially of a mass which moves along an acceleration sensitive axis, and have a pickoff device which detects motion of the mass. Pickoff output resulting from mass motion is fed to a high gain amplifier whose output current flows, in turn, through a force balance coil, producing the force necessary to return the displaced mass to null. This type of an accelerometer, together with its associated amplifier is, in effect, a high-gain, null-seeking servo in which the current flowing through the force balance coil, measured as a voltage across a resistor in series with the coil, is directly proportional to the acceleration applied.

As with gyros, there are many manufacturers of accelerometers. The following listing of equipment characteristics, while by no means complete, is intended to be representative.

Donner Scientific Company

Model	4310
Range	±1/2 to ±35 g or any range totaling 70 g
Voltage output	±7-1/2 v dc
Operating temperature	-40 to +200 F
Linearity	.05 percent of full scale
Threshold	0.001 percent of full scale
Hysteresis	0.02 percent of full scale
Non-repeatability	0.01 percent of full scale
Zero output	0.5 percent of full scale
Cross axis sensitivity	0.002 g/g
Temperature sensitivity	0.001 percent/deg F to 0.05 percent/deg F
Natural frequency	30 to 300 cps
Damping ratio	0.3 to 5
Shock	100 g momentary impact
Vibration	± 25 g 20 to 2000 cps
Input power	± 15 dc ± 15 percent (10 ma max.)
Storage temperature	-65 to +200 F

Kearfott Division, General Precision, Inc.

Type number	425093-1	326778-1	C70 2401 005
Range of measurement	± 5 g	± 10 g	± 20 g
Scale factor	5.000 ma/g		1 v/g
Operating temperature	Performance optimized within any 20 F range between +50 and +160 F		150 ± 10 F
Linearity	Within 0.02 percent of applied acceleration.	Within ± 0.0059 percent of applied acceleration.	5×10^{-6} g/g^2
Threshold	Less than 5×10^{-7}g		2×10^{-7}g
Zero stability	± 0.00005 g day to day; less than ± 0.00002 g over any continuous time interval		0.00001 g
Vibration	± 5 g peak from 20 to 2000 cps		up to ± 20 g peak to 2000 cps
Storage temperature	-45 to +165 F	+60 to 170 F	-65 F to +200 F
Scale vactor variation	± .01 percent randomness		.03 percent/year
Excitation	2.2 v 1600N	3.4 or 6 v 4000N	6 v, 3800 cps
Natural frequency	120 cps	160 cps	220 cps
Frequency response	Flat to 60 cps	Flat to 100 cps	Flat to 250 cps
Shock	2 lb	2 lb	4 ounces

C. COMPUTERS

Because of the versatility required for this application, only digital computers were included in this survey.

An important application of digital computers is the solution of problems involving real-time control. That is, a fixed solution rate is required in at least part of the problem. A central computer for such a system is normally required to carry out vector algebra, integration, differentiation, extrapolation, function generation, limiting, and decision-making.

A digital computer used for real-time control samples the inputs and computes output signals. The time lag between a sensed disturbance and a correcting output must be short enough to provide the desired overall stability. As a separate consideration, the calculation frequency must be high enough to provide the desired dynamic response. Also, the accuracy should be high enough to ensure that the overall system accuracy is not appreciably reduced.

Digital computing techniques may be divided into two categories: (1) whole-value computation and (2) incremental computation. Both techniques are applicable to real-time control problems.

With whole-value techniques, any problem yielding to numerical analysis may be solved. The whole-value computer processes information by performing programed step-by-step arithmetic operations such as addition, multiplication, etc. Most machines of this type have the ability to recognize the sign of a number or the equality of two numbers. This allows the computer to choose one of several sequences of operations to be carried out as a function of some preceding computation. A digital computer that operates in this manner is commonly referred to as a general-purpose computer or GP.

In a GP, entire numbers are transferred from place to place during computation. When one function is being evaluated, the entire computer is dedicated to the job. Each solution made by a GP is basically independent of the previous solution. For example, the sine of an angle is computed by evaluating several terms of a series. However, in a real-time control problem the sine of an angle would probably be required continuously, and a GP would be required to repeatedly evaluate the series.

A digital computer employing the technique of incremental computation by use of integrators is called a digital differential analyzer (DDA or DA). The digital integrator generates an approximation to the equation $dz = ydx$. By interconnecting integrators in some sense, as are analog integrators, it is possible to generate solutions to differential equations of this form.

The DA integrator mechanized the equation $dz = ydx$ as $\Delta z = y \Delta x$. Only increments of the variable x, y, and x are transferred from place to place within the DA. The integrand y is formed by continuously adding Δy increments to the previous integrand value. Figure V-1 shows the mechanization of the equation $\Delta z = y \Delta x$ where y and R are storage registers. The Δz output of an integrator may be used as Δy or Δx inputs to other integrators. All integrators in a DA are processed sequentially through a single computation center. One processing of all integrators is called a major cycle of the DA. Because previous values are held from one major cycle to the next, only updating of previous information must be accomplished by the computation center. Therefore, with a limited amount of hardware, the DA is able to handle many simultaneous problems in an efficient manner.

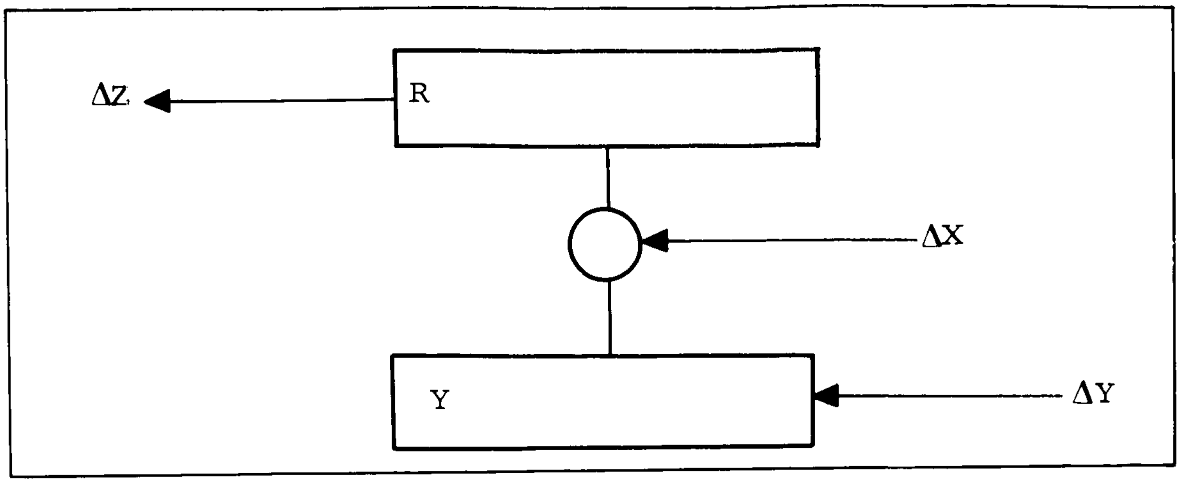

Figure V-1. Integrator Representation of $\Delta z = y \Delta x$

Any integrator in a DA may be operated as a decision element. This gives the DA decision-making capabilities. Decision elements are mechanized in such a manner that the Δz output is the same as the Δx input if the integrand is positive or zero. If the integrand is negative, Δz is zero.

The manner in which the integrators of a DA are interconnected defines a family of curves. The particular member of the family along which extrapolation occurs is determined by the initial conditions. By using decision elements, it is possible to switch from one family to another if both families of curves have been generated. Therefore, it is possible to do all the computations necessary for real-time control using incremental techniques. However, for complex problems involving lengthy control loops and many decisions, the speed and size penalty becomes prohibitive.

It is clear that whole value and incremental machines have their strong and weak points. The potential capabilities of each are strong enough so that no problem that can be solved by one cannot be solved by the other. However, the two techniques are complementary in their mechanization. The great power of the whole value technique (the point solution of a set of equations) is not practical using incremental methods and the forte of incremental methods (efficient extrapolation) is not easily obtainable in any other way.

It would be desirable to have a technique available whereby the conditions at any instant of time for any of the alternate problems could be introduced into the incremental machine so that it could extrapolate along the requisite curve. Such a technique could be effected by a machine which is a combination GP and DA with the computation centers operating distinctly in parallel, requiring no attention from each other for normal computation. However, the GP should have control over the DA when the occasion demands. This can be accomplished if the GP can address the entire DA memory and can, therefore, alter DA data and coding. In this manner, a single machine may be built that operates at moderate repetition rates and utilizes a relatively small amount of circuitry. Each portion of the machine may exhibit extreme simplicity of design and yet the end result is a machine having excellent flexibility and functional capabilities.

The computing system should be able to accept data in both analog and digital form. It should also be able to provide outputs in this form. The internal language format of these computers is digital. To communicate in an analog fashion, some form of electromechanical or electronic conversion equipment must be included in the system. Signal conditioning equipment is necessary with digital data to assure that the pulse train representing this data is legible. The analog signals may come from sensors and go to control equipment. Digital signals may come from an input device such as a tape reader, and go to an output device, such as a tape punch.

There can be a section within the computer, referred to as a buffer, that temporarily stores input and/or output data and allows the computer to perform these operations in a manner which is economical as far as time is concerned.

The tape reader allows the computer access to permanently stored data. The tape punch allows for the creation of a permanent record of data stored within the machine.

Digital computers generally have control panels. The control panel may perform the following functions:

1. Initiate power to the computer
2. Controls computer filling from a tape reader
3. Provides manual filling from the keyboard or typewriter
4. Provides manual control over computer modes of operation
5. Provides manual control over several conditional transfer instructions
6. Provides independent manual control over operation of DA and GP sections

7. Provides manual control over the start of the program

8. Indicates computer states

9. Indicates computer troubles

10. Provides manual and programed readout of words in memory

11. Acts as a programed output device

12. Provides test points from computer

Table V-2 lists the pertinent features of the computers for which replies were received during the survey.

D. CONVERTERS, ADAPTORS, COUPLERS

1. Multiplexer

A multiplexer is used to display several pieces of information so that they can be viewed simultaneously. The multiplexer, functionally, operates as a commutator. It connects each input, sequentially, to the output for a short period of time.

Several commercially available multiplexers are listed below:

1. Astrodata - Anaheim, California

   Model - 956-100 W

   The high level multiplexer is manufactured from a standard line of plug-in cards. High quality, low cost de-operational amplifiers for current summing are included.

   Model - 956-100 W would include:

   a. 4 amplifier cards (8 channels, 2 channels per card)
   b. 1 high level gate card
   c. 1 multiplexer driver card
   d. 1 axis crossing detector card (for triggering on 400-cps voltage)
   e. 1 flip-flop card
   f. 1 logic card

   The approximate cost is $4000.

Table V-2. Features of Digital Computers Surveyed

Manufacturer	Model	Weight	Size	Memory	Access Time	Power	Additional Equipment Necessary	Remarks
Autonetics Div. North American Aviation	Mardan	180 lb	3.05 ft^3	5632 Disk	0.156 usec	825W 120V 400cps 3 ph	Tape Equipment	Has DDA
Computer Control Company, Inc.	DDP-24	1 ton	126.8 ft^3	4096 Core Expandable	3 usec	1400W 115v 60 cps 1 ph	A/D & D/A Equipment	
Control Data Corporation	160A	810 lb	31 ft^3	8192 Core	6.4 usec	26 amps 115 V 60 cps 1 ph	A/D & D A Equipment	
El-Tronics, Inc.	III E		176.5 ft^3	8000/Drum	8 usec	5.5 kw/ 220 v ac	A/D & D/A Equipment	
Epsco, Inc.	275		·40 ft^3	1024 Core Expandable	2 usec		A/D & D/A & Tape Equipment	
General Electric	215	2100 lb	164.8 ft^3 198 ft^3	4096 Core Expandable	36 usec 18 usec	208/120 v 60cps 3 ph 4 wire	A/D & D/A & Console	
H-W Electronics, Inc.	15K	380 lb	33.4 ft^3	4096 Drum	8.3 usec	60cps 1 ph 71 kva 120 v	A/D & D/A Equipment	
Hughes Aircraft Co.	H-330	1 ton	86 ft^3	8192 Core	0.9 usec	5 KW	A/D & D/A Equipment	
Librascope Div. General Precision, Inc.	L-2010 L-90 LPG-21 ASN-24	60 lb 20 lb 155 lb 66 lb	2 ft^3 .3 ft^3 8.7 ft^3 .96 ft^3	4096 Disk 8192 Core 4096 Disk 3584 Drum	5 usec 14 usec 0.39 usec 0.625 usec	500w/400cps /3 ph 30w 300w/110V/ 60 cps 130w	Tape & Console & A/D & D/A Equipment	
Litton Industries	C-900	35 lb	.5 ft^3	4096 Drum	0.23 usec	124 w 60 or 400 cps	Tape, A/D & D/A & Console	Has DDA: welded modules; Std. Package available
Mathatronics, Inc.	Mathatron	50 lb	3.67 ft^3	740 bits/Core	7.5 ms/char.	200 w		
National Cash Register Company	315	1325 lb	75.8 ft^3	6000 digit	6 usec	3 kva 110 v 60cps 1 ph		
Nortronics Div. Northrop Corp.	NDC-100	200 lb	4.2 ft^3	19968 Drum	0.21 usec	2000 w	Tape, A/D & D/A & Console	Has DDA
Pacific Data Systems	PDS 1068			512 Delay Lines	2.5 usec	100 w		Not flyable
Philco Corp	950	900 lb		4096 Core	2.5 usec	400 cps	A/D & D/A Equipment	
	Basic Pac	900 lb	17.5 ft^3	8192 Core	12 usec	115 v 60cps 3 ph	A/D & D/A Equipment	Asynchronous
Scientific Data Systems, Inc.	910 920	600 lb 1000 lb	28.5 ft^3 46.7 ft^3	2048 Core 4096 Core	8 usec	17a 115 v ac 30a 115 v ac	A/D & D/A Equipment	Memory is expandable
Thompson-Ramo-Wooldridge, Inc.	TRW-130	530 lb	10.9 ft^3	8192 Core	6 usec	118 v 60 or 400 cps	A/D & D/A Tape Equipment	

2. United Electro-Dynamics, Inc. - Pasadena, California

   Model - CE-22

   Limited range 0 to +5 v. A 15-channel model is the smallest available, but channels could be paralleled. The main advantage is the small size. The other characteristics are good.

3. Dynatronics, Inc. - Orlando, Florida

   Model - DMM-8 - High level module 8 channel

   0 to 5 v maximum input range.

   The system includes:

   a. 1 Hi-level multiplexer gate
   b. 1 low power multiplexer programer

   The maximum duty cycle of 90 percent is undesirable.

   The maximum on time of 0.5 ms is questionable.

4. Adcole Corporation - Cambridge 39, Massachusetts

   Model - 202 (Modified)

   High level unit modified to 3 poles, 8 channels.

   The limited range is 0 to +5 or 0 to -5 (normal 0 to +5)

2. <u>Miscellaneous Equipment</u>

The following equipment is also required to mechanize the system:

1. Modulator
2. Demodulator
3. AC signal amplifier
4. Phase shifter
5. Filter network
6. Summing network

This type of equipment is generally built using standard circuits for a particular application. No survey was made in this area.

VI. EQUIPMENT RECOMMENDATIONS

The relations between component errors and system errors are given in Appendix III. From these relations, required accuracies of sensors were determined and are listed below. A suggested general specification is included in Appendix VII. Recommendations for particular hardware are made. The hardware recommended will allow the mechanization of the display as developed by the Ames Research Laboratory.

A. POSITION AND RATE SENSORS

Referring to Section III, the pilot should be able to have knowledge of his lateral position to within 15 ft while on the runway. The sensor used must be at least this accurate. As shown in Section V, after a period of time, an inertial navigator cannot deliver this accuracy. In addition, an accuracy of ±50 ft in the position along the runway centerline should be obtained. An altitude accuracy of ±1.5 ft while over the runway is necessary. During the approach, the accuracy requirements are not as stringent, and may be inferred from the system requirements given in Section III.

The accuracy required in rate sensing is ±1 ft/sec when transformed to y and z body rates. Overall rate accuracies of ±2 ft/sec are allowable.

Additional features which would be desirable are that accurate distance measurements are made irrespective of the location of the ground reference point relative to the aircraft, and that the output of the sensor be amenable to the input of the computer so that little or no interface equipment is necessary.

Considering radar, the transformation from slant range to inertial coordinates, in matrix rotation, is

$$\begin{bmatrix} X_I \\ Y_I \\ Z_I \end{bmatrix} = \begin{bmatrix} \psi \end{bmatrix} \begin{bmatrix} \theta \end{bmatrix} \begin{bmatrix} \phi \end{bmatrix} \begin{bmatrix} \zeta \end{bmatrix} \begin{bmatrix} \eta \end{bmatrix} \begin{bmatrix} R \\ 0 \\ 0 \end{bmatrix}$$

if a Z-Y-X gimbal arrangement exists for the gyros and a Z-Y gimbal arrangement exists for the radar antenna. The most critical of the position specifications is the one concerning lateral displacement on the runway (see Section III). An expression relating the lateral displacement caused by different sensor errors is derived in Appendix III and approximate values of these errors are plotted. It can be seen in Appendix III, Figure AIII-2 that the approximate lateral sensing error at a point nominally near touchdown (X_I = 10,000 ft, h = 100 ft) is about 140 ft. This is an order of magnitude greater than the tolerable error. Thus, radar does not seem to be accurate enough for this application. It can also be seen in the appendix and by the discussion in Section V, that the radar has a limited field of view.

Of the two types of DME, the pulsed type, which is similar to radar, cannot be considered since it is less accurate than radar. The curves shown in Appendix III are based on the Aeris system manufactured by the Cubic Corporation. Figure AIII-6 shows the errors that can be expected in the two horizontal components of position and in altitude. It should be noted that errors in both X_I and Y_I are quite small, specifically the error in Y_I never exceeding 10 ft anywhere on the runway, although the error in the altitude becomes quite large.

Altitude may be measured by any one of several means. Barometric altimeters are generally accurate to within 50 ft of altitude. For this application, this type of instrument could not be used at altitudes below about 350 ft.

Another type of altitude sensor is the radar altimeter. Those which have sufficient accuracy, in the order of 1 to 2 ft, generally are of limited range. The maximum altitude which can be measured is about 500 to 1000 ft. Radar altimeters, however, measure terrain clearance rather than altitude above the runway. This is a desirable feature near the ground. However, during the approach, the terrain surrounding the airfield is generally not at the same elevation as the runway. Above some altitude, nominally several hundred feet, the information sensed by this means is no longer meaningful as a measure of elevation above the runway.

A combination of barometric and radar altitude sensors should provide satisfactory results.

The sensing of rate can be accomplished by using any of the above position sensors. Rate terms are available from each. In the case of the inertial navigator, the first integral of the accelerometer outputs would be velocity. Rate is available from the radar and DME where it is generated to mechanize range tracking loops. In this case, the rate signals will probably be noisy as a result of generation by differentiation. In order to obtain good high frequency response to aircraft velocity, and to eliminate the noise content of these signals, it has been found through experience that by using these noisy rate signals in conjunction with an accelerometer and some filtering, a good rate signal may be obtained. These statements also apply to the case where a computed position is used to derive rate. Again, if DME is used, a radar altimeter and a barometric altitude rate sensor would be needed.

The advantages and disadvantages of the different sensors are listed in Table VI-1.

Table VI-1. Sensor Advantages and Disadvantages

Sensor	Advantages	Disadvantages
Inertial Navigator	Measures distances in convenient coordinate system. Can use potentiometer pick-offs for angles. Does not need ancillary rate sensors. Total integrated subsystem. Available to operate with analog or digital computer. Highly accurate angular measurements. Versatile; can be used for other missions.	Position signals deteriorate too rapidly to be useful. Extremely expensive. Involved alinement procedure.
Radar	Versatile; can be used for other missions. Contains a display which with minor modifications can be used for this application.	Does not have required accuracy laterally or in altitude. Needs operator. May need ground based beacon. Needs ancillary rate and angular sensors. Limited field of vision.

Table VI-1. (Cont)

Sensor	Advantages	Disadvantages
CW/DME	Meets accuracy in horizontal plane. Can be used for navigation in neighborhood of airport. Measures range in any direction. May have capability of voice transmissions.	Needs ground transponders. Needs ancillary rate and angle sensors. Accurate output in digital form only.

From this it can be seen that although it is not perfect, the CW/DME is the best of the three types of sensors reviewed for this application. In reviewing the component survey (Section V), the characteristics of the two CW/DME considered can be compared. It is recommended that the Aeris CW/DME manufactured by the Cubic Corporation be used for this application. This equipment is more applicable than its competitors because (1) only one airborne transceiver is required, and this transceiver can interrogate up to four ground based transponders, and (2) its accuracy is better.

Accelerometers, a radar altimeter, a barometric altimeter, and a barometric rate of climb indicator with electrical pickoffs are required for DME. Standard autopilot instruments will be suitable for the barometric altimeter and rate of climb indicator. These are commercially available, and are not critical sensors so that any one of several could be used.

The accelerometers used should be of the force balance type because of the accuracy requirements. It would be desirable to have an output which is compatible with the input of the computer used. In a later portion of this section, the MARDAN computer will be recommended. It accepts d-c inputs whose voltage swing is ±10 v. For this reason, the Donner accelerometer model 4310 is recommended although any similar instrument might be used. These should have the following minimum characteristics:

1. Nonlinearity. Less than 0.05 percent of full scale
2. Hysteresis. Less than 0.02 percent of full scale

3. Resolution. Better than 0.0001 percent of full scale
4. Repeatability. Better than 0.01 percent of full scale
5. Zero output. Less than 0.05 percent of full scale
6. Cross axis sensitivity. Less than 0.002 g/g referred to true sensitivity
7. Temperature sensitivity. Less than 0.005 percent/F
8. Damping ratio. 0.7 ± 0.1
9. Range. ±1 g about nominal (provision for 1 g)
10. Output voltage. ±7-1/2 v dc standardized between instruments
11. Case alinement to true sensitive axis. ±1/4 deg

The radar altimeter should be able to sense both altitude and altitude rate accurately. It should operate satisfactorily up to at least 500 ft. The altitude resolution should be at least 0.5 ft., while the accuracy should be at least 1.5 ft at touchdown. The range in altitude rate should be at least ±50 ft/sec while the rate accuracy at touchdown should be 1.5 ft/sec or better. Both outputs should be linear. The radar altimeters surveyed are listed in Section V. The Honeywell model 7182 is recommended because it meets the requirements, and is one of the very few radar altimeters that have passed the Federal Aviation Agency (FAA) flight evaluation for landing system application.

B. ANGLE SENSORS

The angle sensors are described in Section V. Table VI-2 compares the various types of sensors.

Table VI-2. Types of Sensors - Advantages and Disadvantages

Sensor	Advantages	Disadvantages
Magnetic Compass and Vertical Gyro	Simplicity Low cost No long term drift	Heading only accurate during steady state. Affected by changes in magnetic fields.

Table VI-2. (Cont)

Sensor	Advantages	Disadvantages
Directional Gyro and Vertical Gyro	Good dynamic response in heading while maneuvering Not affected by changes in magnetic fields Accuracy can be improved by cross torquing	Long term drift may present a problem. May lose heading accuracy at large bank angles.
Two gyro reference	Good dynamic response in heading while maneuvering. Not affected by changes in magnetic fields. Keeps heading accuracy at large bank angles. Can have potentiometer outputs.	Long term drift may present a problem.
North seeking gyro and vertical gyro	Good dynamic response in heading while maneuvering. Not affected by changes in magnetic fields. Keeps heading accuracy at large bank angels. No long term drift.	Complex system. Needs accurate velocity information.
Magnetic compass, directional gyro, and vertical gyro	Good dynamic response in heading while maneuvering. No long term drift. Accuracy can be improved by cross torquing.	May lose heading accuracy at large bank angles. Overall steady state heading accuracy dependent on magnetic compass. Complex system.

Table VI-2. (Cont)

Sensor	Advantages	Disadvantages
Magnetic compass and two gyro reference platform system	Good dynamic response in heading while maneuvering. Keeps heading accuracy at large bank angles. No long term drift. Can have potentiometer outputs	Overall steady state heading accuracy dependent on magnetic compass. Complex system.

The requirements on the angle sensors are that they have an accuracy of ±0.25 deg over the period of the flight and have a linear output. The sensors must be capable of operation over the range of ±85 deg in roll and ±45 deg in pitch. It would also be desirable to have the outputs compatible with the computer.

It is felt that any of the systems on the above list, with the exception of the first, would be acceptable for this application. The two gyro reference platform is recommended as a system which is adequate and not excessively complex. However, because of considerations relative to the general use of the test vehicle and beyond the scope of this study, it might be desirable to use some other attitude reference system. On the basis of system requirements only, no one system is preponderantly superior for this application.

C. COMPUTER

The computer recommended is the MARDAN computer. It meets the general specification, and the detailed computer specification listed in Appendix VII. It is manufactured by the Autonetics Division of North American Aviation.

D. MULTIPLEXER

The original display contains six symbols. As many as nine were used during Autonetics simulation. A minimum of eight, three pole multiplexer channels (3 synchronized, eight channel multiplexers) should be used. More channels might be useful for testing the use of additional displayed information.

The multiplexer should be a high level device. The voltage levels of the inputs should be similar to the outputs of the computer and the multiplexer outputs should be sufficient to drive the display. The multiplexer should not load down the computer outputs. It should have the following characteristics:

Minimum sampling	400 channels per second
Signal duty cycle (Ratio of channel on-time to channel period)	$\sim$ 100 percent
Signal range	0 ± 10 v
Input impedance* Looking into "On" gate	1 meg
Output impedance	$<$ 10 k
Linearity	$\geq$ 0.25 percent
Noise	$<$ 30 mv
Switching time	$<$ 10 usec
Voltage requirements (desired)	+28v dc

* Without input summing amplifiers. Less than 2 k with summing amplifiers.

It is recommended that the model 956-100W multiplexer manufactured by Astrodata, Anaheim, California, or its equivalent be used. This would consist of:

4 amplifier cards (8 channels, 2 channels per card)

1 high level gate card (24 channels)

1 multiplexer driver card

1 axis crossing detector card (for triggering on 400 cps voltage)

1 flip-flop card

1 logic card

It was chosen because of its modular design, it is a high-level multiplexer manufactured from a standard line of plug-in cards, and has a high quality d-c operational amplifier for current summing.

E. MISCELLANEOUS

The specific types of interface equipment and the requirements are dependent on the particular combination of sensors, computers, and displays which are used. The requirements for interface equipment that are other than standard type coupling (isolation and scaling amplifiers, etc.) are discussed in other parts of this report. The type of equipment that might be necessary between the computer and the sensors is listed in Section VII and that at the input to the display in Appendix IV.

VII. SYSTEM RECOMMENDED

A. GENERAL

A block diagram of the recommended system is presented in Figure VII-1. The system utilizes DME as the dominant sensor. Equations describing this configuration are given in Appendix I.

B. OPERATION

Operational procedure for a flight test version of the system installed in an aircraft is as follows:

1. Turn on aircraft a-c and d-c power supplies.

2. Turn on all manual blind landing system equipment.

3. Perform self-test operations for the following equipment:

   a. radar altimeter
   b. DME
   c. digital computer

4. Manually set in the landing runway parameters ψ_o, a, b, and ℓ (digital computer). For test purposes these may be included in the program tape.

5. Set the pressure altimeter for existing conditions.

6. When airborne and in vicinity of landing runway, perform approach and landing maneuvers as discussed in Appendix II. Use of displayed range dot, heading marker, and runway centerline presupposes modification of the display as developed by Ames.

C. ADDITIONAL USES

Inasmuch as the major emphasis is on the approach and landing phase of the general problem, this phase was used to determine the equipment requirements. The only consideration given to the takeoff, go-around and holding pattern maneuvers relative to equipment requirements was that the sensors be functional during all of these maneuvers.

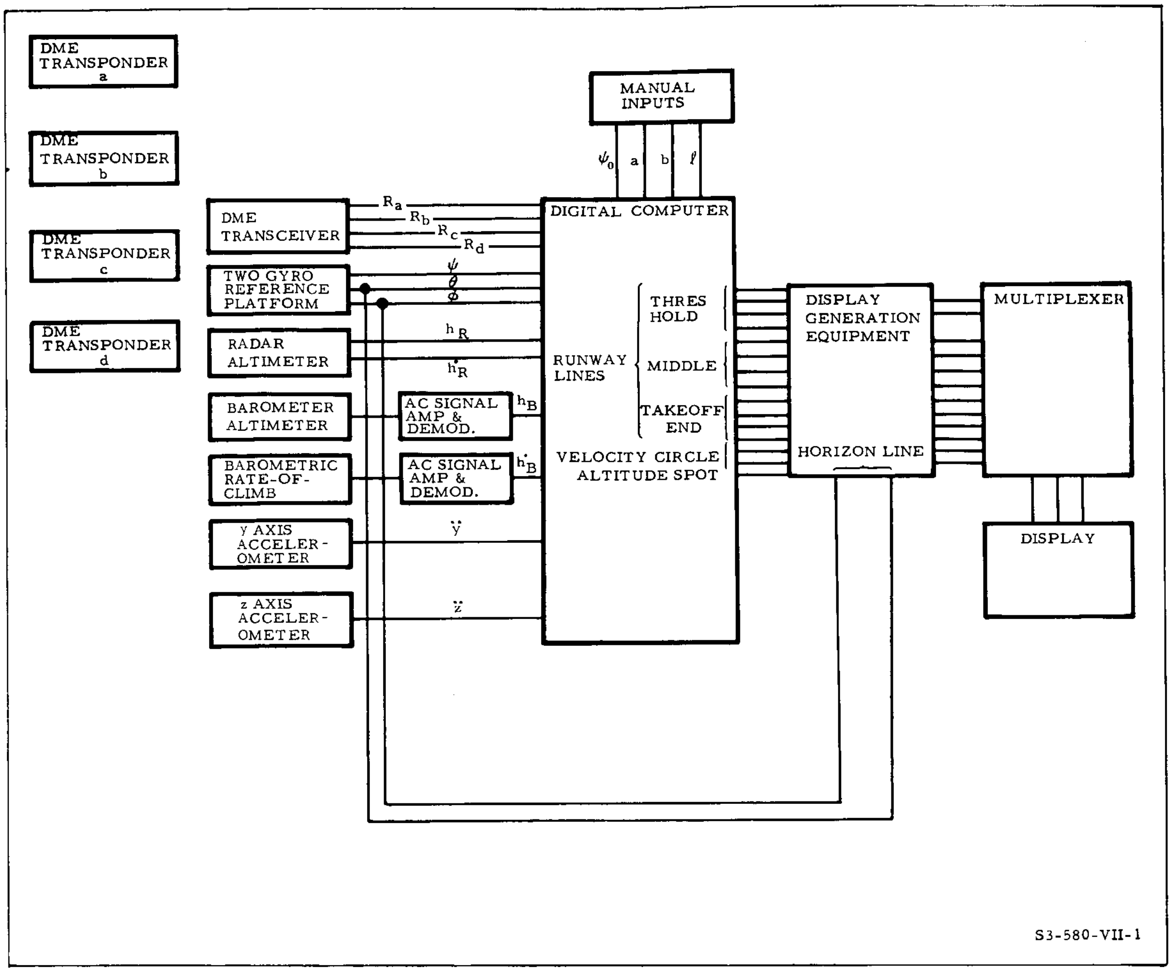

Figure VII-1. The Recommended System

For example, forward looking radar would be ruled out because of its limited antenna motion which would make it essentially inoperable except when the aircraft is flying generally toward the ground target (near or on the airport).

The display, while most appropriate for the approach and landing has in its original form little information of use for other maneuvers. The original display contained a horizon line, three lines representing a runway, a circle representing the total velocity vector and a spot indicating the last 100 ft of altitude above the ground. For a takeoff or the initial portion of a go-around all of this information would still be available on the display except the runway lines. (The "far-end" line would remain for a while but its usefulness is doubtful.)

During a circular holding pattern the horizon line and the velocity circle would be continuously available and the runway lines might periodically be visible on the display depending on the particular pattern being flown.

Although, with the use of the normal flight instruments in conjunction with the display, a takeoff might be accomplished, the original display format is of no value for flying a holding pattern.

As indicated in Appendix II the presentation of some additional information enhances the usefulness of the display even for approach and landing. Depending upon the particular set of equations solved by the computer, the complete position, velocity, and attitude of the aircraft relative to the runway are available in the computer. With the addition of an indicator on the periphery of the display showing range to the runway, relative hearing of the runway centerline, and a line from the aircraft to a point on the runway, the display may be used for navigation within the range of the DME. An indication of the lateral displacement from the extention of the runway centerline and heading relative to the runway would be very helpful. The addition of a runway centerline extended to the horizon would also be useful.

To facilitate the use of the system consider a flight plan consisting of a takeoff into a holding pattern and subsequent return to the field.

During the takeoff roll, lateral directional information is available on the display both as the runway centerline display and as bearing to the far end of the runway. In addition the lateral displacement and relative heading are available on supplementary instruments. The runway centerline is by far the most useful of these for takeoff roll. Position along the runway is provided by the displays peripheral indication of range to the far end of the runway and to some extent by the "far-end" runway line. Ground speed is available in the computer but not displayed. No airspeed or stall warning is directly available. However, angle of attack can be seen on the display as proportional to the distance between the velocity circle and the aircraft symbol.

After takeoff the altitude (terrain clearance) is available on the display to an altitude of 100 ft. A constant climb angle, γ , can be established by noting the distance on the display between the velocity circle and the horizon line, which is proportional to γ. Directional control can be maintained as before, except that the runway centerline may no longer be in sight. For this purpose the runway centerline may be displayed as extending past the limits of the runway.

The desired climb angle may be maintained until the altitude of the holding pattern is reached as noted on the aircraft's barometric altimeter. A circular holding pattern circumscribing the airport may be acquired by continuing on the extention of the runway centerline until the range indication on the display indicates the desired radius of the pattern. The pilot may then bank the aircraft, right or left, to maintain the range constant. Any number of orbits may be made. To complete the flight plan the approach pattern may be acquired in either of two ways, depending on whether the orbiting radius is greater or less than the desired range to start the final approach.

If the radius is greater than the desired range, the pilot should note the bearing angle on the display and as it approaches zero (indicating the aircraft is approaching the runway centerline extention on the approach end) turn toward the runway heading and then proceed as described for an approach.

In the event an approach longer than the radius of the pattern is desired, the following procedure may be used. After the aircraft passes the takeoff end of the runway (bearing 180 deg) the pilot should note the aircraft heading as the bearing reaches 270 or 90 deg depending on the direction of rotation. By rolling out and maintaining the heading existing at the 270- or 90-deg bearing point, or maintaining a constant Y_I as displayed on a meter, a downwind leg parallel to the runway may be flown until the desired range is reached. By heading and turning to maintain this new range, the procedure described in the preceding paragraph may be followed to acquire the final lineup. Many variations of these procedures are possible using the same information described above.

D. FLIGHT TEST RECOMMENDATIONS

It is recommended that, as a minimum, the following tests be made preliminary to demonstrating or evaluating the recommended system. A recommendation of the data that should be obtained is also given.

In order to use the assemblage of equipment to evaluate the display or any one portion of the system, it is necessary that accurate calibration of each individual item be made under the conditions that exist when the equipment is installed and operating in the test environment.

The equipment calibrations required may be divided into four groups: sensing equipment, computing equipment, interface equipment, and displays.

With all of the equipment installed and operating on ship's power, ground tests may be performed to determine that all of the equipment is functioning. In addition, calibration and scaling may be performed on all of the electronic devices except the sensors. Static calibration of the sensors may be performed on the ground, but dynamic airborne calibration is required to obtain the necessary accuracy to use the sensors as standards.

The sensors used to determine position and position rate should be calibrated against a photo-theodolite. The calibration tests should include fly-bys, letdowns, and climbs. The normally limited coverage of photo-theodolite installations may require some other means of calibrating the DME for navigational operation away from the immediate area of the airport. It is recommended that the position and rate outputs of the several sensors be recorded on an airborne oscillograph recorder. Additional parameters that should be recorded for all tests include the aircraft attitude (pitch, roll, and yaw). It is desirable to record as many parameters as possible which describe the aircraft and the equipment within it. Pickoffs should be provided to indicate surface positions, control column positions, flap motion, etc. If a particular parameter is available from more than one sensor, both should be recorded. Time correlation between airborne recordings and ground photographs (photo-theodolite) may be provided by a high intensity flashing light on the outside of the aircraft in view of the camera, the flashes being coded and recorded on the airborne oscillograph.

Inasmuch as a transport type aircraft is assumed to be used for tests and technical observers will be aboard, it is recommended that an "instant print" attachment be provided on the oscillograph. This attachment allows for examination of the data by an observer within a few seconds, and greatly facilitates testing by allowing decisions to be made based on recorded data while still airborne.

An indication of touchdown is required. A switch may be provided on the landing gear to sense landing strut compression, or a tachometer may be installed on one or more of the wheels. From past experience, although these means are desirable (and necessary for a discrete electrical indication of touchdown), it is usually very evident on the oscillograph data when first contact with the runway occurs. If the scaling is sufficiently fine, the roll angle is an excellent indication of touchdown. Aircraft rarely land precisely level, completely balanced in roll. The slight roll moment produced as one gear picks up more of the aircraft weight than the other is quite evident on the roll gyro output. Frequently, it is possible to determine the time of contact of each of the main gear from the data using this indication. Accelerometers have not proved particularly useful for this indication.

It is recommended that the preliminary calibrations of all the equipment be obtained and the raw flight test data be thoroughly examined by technical personnel familiar with the equipment and the particular tests being made. Many minor effects which might be significant in later evaluations are readily apparent in the raw data, but may be completely lost through some data reduction techniques.

VIII. CONCLUSIONS

It has been determined that a system as described in section III can be implemented. The system recommended will be capable of sensing information, operating on it for presentation to the pilot enabling him to take off from an airport, navigate in the immediate vicinity, and make approaches and landings without visual contact with the ground. It was not found to be practical, within the ground rules, to have the entire system airborne. The ground based portion can be limited to portable battery-operated transponders.

It was found that the equipment required for the recommended system, which was not available in production in exactly the required form, could be made applicable with minor modification. No special couplers were required which were not commercially available or could not be built using standard techniques and components.

The equipment necessary to perform these tasks are constant wave frequency modulated DME, a two-gyro reference system, radar and barometric altimeters, a barometric rate of climb indicator, a digital computer, a multiplexer, a cathode ray tube type display, and necessary miscellaneous equipment. The miscellaneous equipment consists of modulators, demodulators, a-c signal amplifiers, scaling and shaping networks, and an oscillator.

It was determined by simulation that the display suggested by Ames Research Laboratory could be used successfully to land an aircraft. Addition and modification to the format improved its usefulness. These additions consisted of a runway centerline and an indication of runway bearing and range. The display provides an excellent source of information for landing and short-range navigational tasks. Glide slope could easily be set up and maintained. Pilots and non-pilots adjust readily to the display.

APPENDIX I. EQUATIONS SUMMARY

This appendix summarizes the equations used in this report. These equations are divided into two categories: those associated with the sensors, and those associated with the display. Equations applicable to the error analysis are contained in Appendix III.

A. SENSORS

Equations were derived to convert both radar and DME measured parameters to inertial coordinates. Because these derivations are straightforward, only the resulting equations are given.

1. Radar

The equations expressing the transformation from sensed parameters to inertial coordinates are best expressed in matrix form. They are

$$\begin{bmatrix} X_I \\ Y_I \\ Z_I \end{bmatrix} = \begin{bmatrix} \\ \psi \\ \\ \end{bmatrix} \begin{bmatrix} \\ \theta \\ \\ \end{bmatrix} \begin{bmatrix} \\ \phi \\ \\ \end{bmatrix} \begin{bmatrix} \\ \zeta \\ \\ \end{bmatrix} \begin{bmatrix} \\ \eta \\ \\ \end{bmatrix} \begin{bmatrix} R \\ O \\ O \end{bmatrix}$$

This matrix is expanded in Appendix III. The assumptions made are that the order of gyro gimbal rotation is Z - Y - X, and that the order of antenna gimbaling is Z - Y.

2. DME

The transponder geometry considered is shown in Figure AI-1. The equations which transform the range information to inertial information are

Two transponders plus altimeter

$$X_A = \pm \left[\left(\frac{R_a^2 + R_b^2}{2} \right) - m^2 - Y_A^2 - (h - h_t)^2 \right]^{\frac{1}{2}} \pm \ell$$

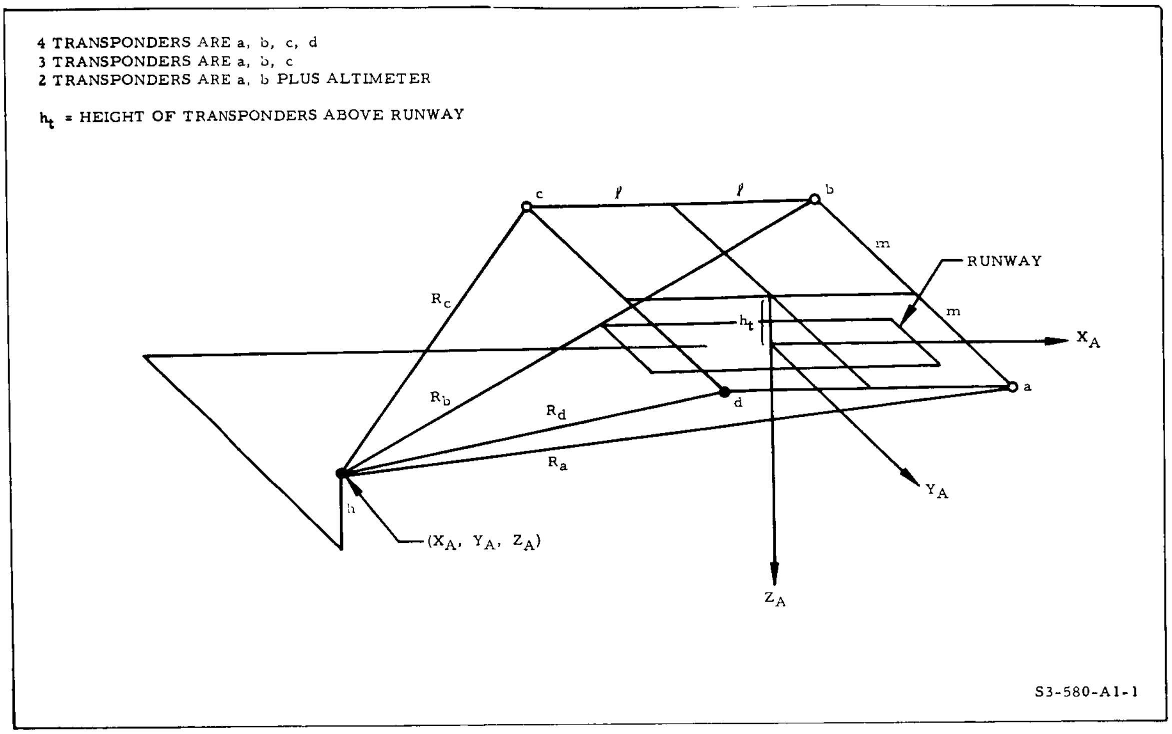

Figure AI-1. Transponder Geometries

$$Y_A = \frac{R_b^{\,2} - R_a^{\,2}}{4m}$$

$$Z_A = -h$$

Approximate solution for X

$$X_A \approx \mp\left(\frac{R_a + R_b}{2}\right) \mp \ell$$

$$\text{if } \frac{1}{2}\left[\left(\frac{Y_A}{X_A - \ell}\right)^2 + \left(\frac{Z_A + h_t}{X_A - \ell}\right)^2 + \left(\frac{m}{X_A - \ell}\right)^2\right] \ll 1, \text{ of order } 0.1$$

where the sign of X_A is the same as the sign of $\dot{R}_a + \dot{R}_b$

Three transponders $X_A = \dfrac{R_c^2 - R_b^2}{4\ell}$

$$Y_A = \frac{R_b^2 - R_a^2}{4m}$$

$$Z_A = -\left[\left(\frac{R_a^2 + R_c^2}{2}\right) - (\ell^2 + m^2) - X_A^2 - Y_A^2\right]^{\frac{1}{2}} - h_t$$

Four transponders plus two altimeters

$$X_A = \frac{1}{2}\left[\left(\frac{R_c^2 - R_b^2}{4\ell}\right) + \left(\frac{R_d^2 - R_a^2}{4\ell}\right)\right] \qquad \text{(AI-1)}$$

$$Y_A = \frac{1}{2}\left[\left(\frac{R_c^2 - R_d^2}{4m}\right) + \left(\frac{R_b^2 - R_a^2}{4m}\right)\right] \qquad \text{(AI-2)}$$

$$Z_A = -\frac{1}{2}\left[h_1 + h_2\right] \qquad \text{(AI-3)}$$

Use of Equations AI-1, AI-2, and AI-3 is recommended. In addition, for the four transponder case, another set of geometry was investigated (see Figure AI-2). The equations for this case are

Four transponders

$$X_A = \frac{R_c^2 - R_b^2}{4\ell}$$

$$Y_A = \frac{R_b^2 - R_a^2}{4m}$$

$$Z_A = \frac{R_d^2 - R_c^2}{2n} - \frac{h}{2} - h_t$$

3. Accelerometers

When the radar altimeter is used to measure altitude rate, its output should be filtered as follows:

$$\dot{h} = \frac{\left(\frac{K_2 \beta}{S + \beta}\right)\ddot{h}_I + \left(\frac{K_1 \alpha}{S + \alpha}\right)\left(\frac{K_2 \beta}{S + \beta}\right)\dot{h}_R}{1 + K_3\left(\frac{K_1 \alpha}{S + \alpha}\right)\left(\frac{K_2 \beta}{S + \beta}\right)}$$

where $\dot{h}$ is the altitude rate signal to be used as $\frac{dZ_A}{dX_A}$ when the radar altimeter is being used as the primary altitude rate sensor. The output of the z body accelerometer is $\ddot{h}_I$. Nominal values of the constants are

$$K_1 = 1$$
$$K_2 = 15$$
$$K_3 = 1$$
$$\alpha = 1.5$$
$$\beta = 0.15$$

and S is the Laplace transform operator. The constants given may have to be changed slightly in order to fit the exact dynamics of the particular sensors.

In addition, it may prove necessary to perform a similar operation $\frac{dZ_A}{dX_A}$ when the radar altimeter is not being used. This also

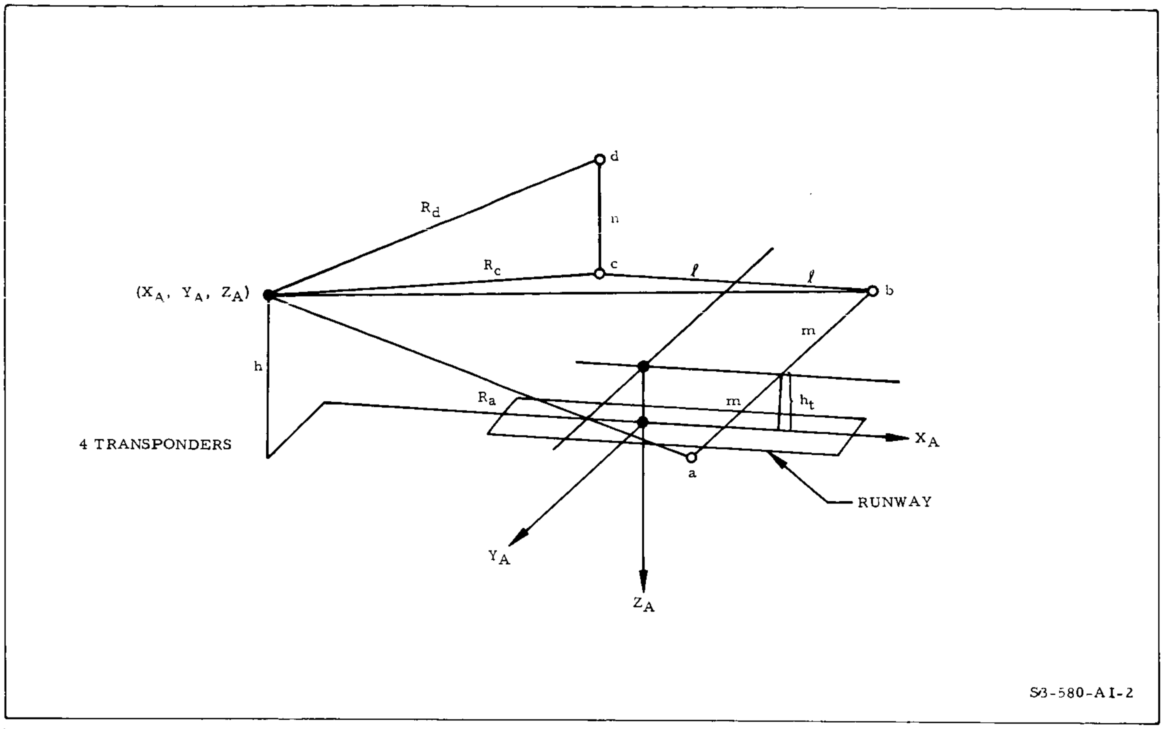

Figure AI-2. Four Transponder Geometry

applies to $\frac{dY_A}{dX_A}$. In these cases, the rates derived from DME and the barometric rate of climb meter would be analogous to $\dot{h}_R$

B. DISPLAY

Several sets of equations were derived for generating the display. Only the derivation of the recommended set is included. The other set of equations is included for information only. The use of small angle approximations will be acceptable for the approach and landing.

The velocity circle generated by the approximate equations of the recommended set has some limitations on it. This can be seen by referring to Appendix III.

1. Recommended Set

Figure AI-3 defines the geometry used. From this figure, it can be seen that

$$(\bar{R}_i - \bar{R}_A) = (X_i - X_A)\,\bar{I} + (Y_i - Y_A)\,\bar{J} + (Z_i - Z_A)\,\bar{K} \qquad \text{(AI-4)}$$

$$d\bar{R}_A = dX_A\,\bar{I} + dY_A\,\bar{J} + dZ_A\,\bar{K} \qquad \text{(AI-5)}$$

The unit vectors $\bar{I}$, $\bar{J}$ and $\bar{K}$ can be expressed in the body axis unit vectors $\bar{i}$, $\bar{j}$ and $\bar{k}$ as follows

$$\bar{I} = \bar{i}\left[c\psi c\theta\right] + \left[c\psi s\theta s\phi - s\psi c\phi\right] + \bar{k}\left[c\psi s\theta c\phi + s\psi s\phi\right] \qquad \text{(AI-6)}$$

$$\bar{J} = \bar{i}\left[s\psi c\theta\right] + \bar{j}\left[s\psi s\theta s\phi + c\psi c\phi\right] + \bar{k}\left[s\psi s\theta c\phi - c\psi s\phi\right] \qquad \text{(AI-7)}$$

$$\bar{K} = \bar{i}\left[-s\theta\right] + \bar{j}\left[c\theta s\phi\right] \qquad \text{(AI-8)}$$

where: $c\psi = \cos\psi$, $s\,\psi = \sin\psi$, etc.

assuming a Z - Y - X rotation from inertial to body axes.

The display is in the $\bar{j}$, $\bar{k}$ plane. Thus the $\bar{j}$ components of the geometry are proportional to the horizontal deflection signal, and the $\bar{k}$ components are proportional to the vertical deflection.

The coordinates of the display are the y and z body coordinates. Combining Equations AI-4, AI-6, AI-7, and AI-8, it can be seen that

$$\left(\frac{y_D}{R}\right)_i = \Big\{ (X_i - X_A)\left[c\psi s\theta s\phi - s\psi c\phi\right] + (Y_i - Y_A)\left[s\psi s\theta s\phi + c\psi c\phi\right]$$

$$+ (Z_i - Z_A)\left[c\theta s\phi\right]\Big\}\left(\frac{1}{R_i}\right), \qquad (X_i - X_A) > 0$$

$$= \pm 1, \qquad (X_i - X_A) < 0$$

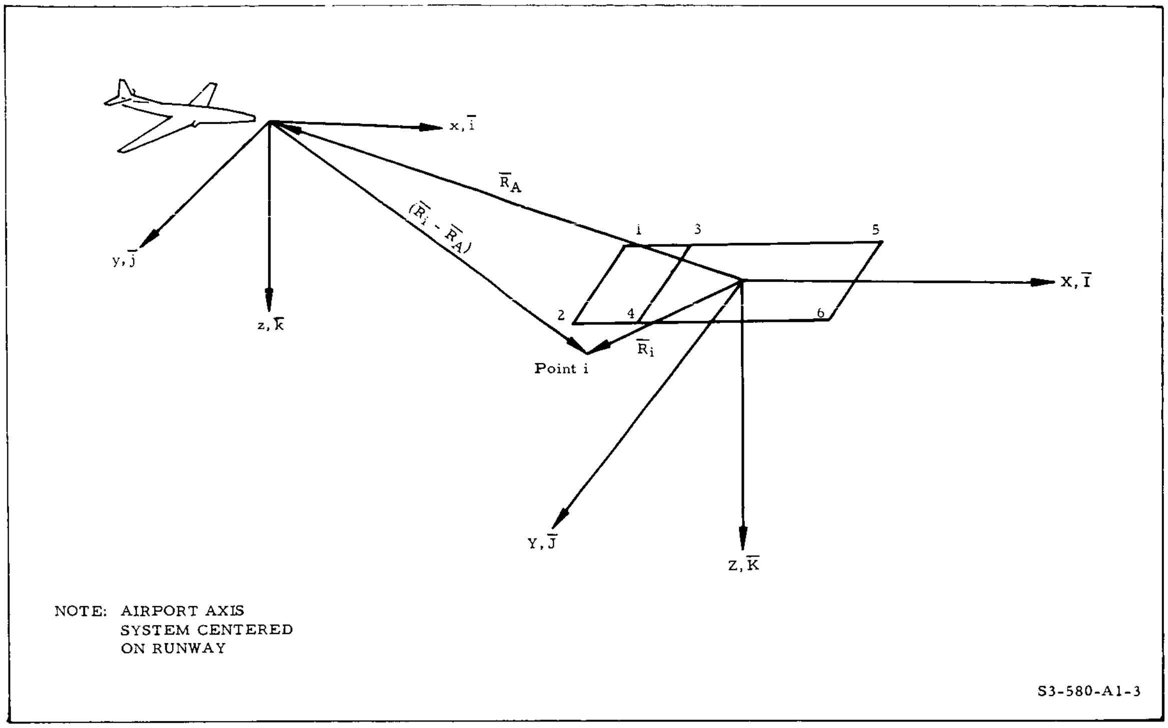

Figure AI-3. Axes Systems

$$\left(\frac{z_D}{R}\right)_i = \left\{ (X_i - X_A)\left[c\psi s\theta c\phi + s\psi s\phi\right] + (Y_i - Y_A)\left[s\psi s\theta c\phi - c\psi s\phi\right] + (Z_i - Z_A)\left[c\theta c\phi\right]\right\}\left(\frac{1}{R_i}\right), \qquad (X_i - X_A) > 0$$

$$= \pm 1, \qquad (X_i - X_A) < 0$$

$$R_i = \left[(X_i - X_A)^2 + (Y_i - Y_A)^2 + (Z_i - Z_A)^2 \right]^{\frac{1}{2}}$$

Notes: 1. $c\psi = \cos\psi$, $s\psi = \sin\psi$, etc.
2. i = 1, 2; 3, 4; 5, 6; locating end points of runway lines.

Small angle approximations were made and the following set of equations resulted.

$$\left(\frac{y_D}{R}\right)_i \approx \left[\left(\frac{Y_i - Y_A}{X_i - X_A}\right) - \psi\right] + \left[\theta + \left(\frac{Z_i - Z_A}{X_i - X_A}\right)\right]\phi, \quad (X_i - X_A) > \qquad \text{(AI-9)}$$

$$\pm 1, \qquad (X_i - X_A) < \qquad \text{(AI-10)}$$

$$\left(\frac{z_D}{R}\right)_i \approx \left[\theta + \left(\frac{Z_i - Z_A}{X_i - X_A}\right)\right] - \left[\left(\frac{Y_i - Y_A}{X_i - X_A}\right) - \psi\right]\phi, \quad (X_i - X_A) > \qquad \text{(AI-11)}$$

$$= \pm 1, \qquad (X_i - X_A) < \qquad \text{(AI-12)}$$

Notes: 1. $\Sigma = \frac{1}{2}\left[\left(\frac{Y_i - Y_A}{X_i - X_A}\right)^2 + \left(\frac{Z_i - Z_A}{X_i - X_A}\right)^2\right]_{max.}$ of order 0.1

2. ψ, θ "small" (of order 0.1)

ϕ "not so small" (of order 0.3)

3. i = 1, 2; 3, 4; 5, 6; locating and points of three runway lines

Equations AI-9, AI-10, AI-11, and AI-12 are recommended for use to drive the runway line display.

The equations necessary to drive the velocity circle are derived from AI-2, AI-3, AI-4, and AI-5. They are

$$\left(\frac{y_D}{R}\right)_V = \left\{ dX_A \left[c\psi\, s\theta\, s\phi - s\psi\, c\phi\right] + dY_A \left[s\psi\, s\theta\, s\phi + c\psi\, c\phi\right] + dZ_A \left[c\theta\, s\phi\right]\right\} \frac{1}{\left|d\bar{R}_A\right|} \qquad \text{(AI-13)}$$

$$\left(\frac{z_D}{R}\right)_V = \left\{ dX_A \left[c\psi\, s\theta\, c\phi + s\psi\, s\phi\right] + dY_A \left[s\psi\, s\theta\, c\phi - c\psi\, s\phi\right] + dZ_A \left[c\theta\, c\phi\right]\right\} \frac{1}{\left|d\bar{R}_A\right|} \qquad \text{(AI-14)}$$

$$|d\bar{R}_A| = \left[(d X_A)^2 + (dY_A)^2 + (dZ_A)^2 \right]^{1/2} \qquad \text{(AI-15)}$$

Note: $dX_A \Rightarrow X_A(t) - X_A(t - \tau)$ etc, where τ is some differencing time interval small compared to airframe response time.

Several sets of equations arise after small angle approximations are made. The first is

$$\left(\frac{y_D}{R}\right)_V \approx \left[\left(\frac{dY_A}{dX_A}\right) - \psi \right] + \left[\theta + \left(\frac{dZ_A}{dX_A}\right) \right] \phi$$

$$\left(\frac{z_D}{R}\right)_V \approx \left[\theta + \left(\frac{dZ_A}{dX_A}\right) \right] - \left[\left(\frac{dY_A}{dX_A}\right) - \psi \right] \phi$$

Notes: 1. $\Sigma = \frac{1}{2} \left[\left(\frac{dY_A}{dX_A}\right)^2 + \left(\frac{dZ_A}{dX_A}\right)^2 \right]_{max} \ll 1$, of order 0.1

2. ψ, θ "small", (of order 0.1)

ϕ "not so small", (of order 0.3)

3. Improved full range accuracy can be achieved by using $(1 + \frac{\Sigma}{2})(dX_A)$ for divisor.

An improved set is

$$\left(\frac{y_D}{R}\right)_V \approx \left\{ \frac{dY_A}{dX_A} \left[1 - \frac{1}{4} \left(\frac{dY_A}{dX_A}\right)^2 \right] - \psi \right\} + \phi \left\{ \frac{dZ_A}{dX_A} \left[1 - 0.45 \left(\frac{dY_A}{dX_A}\right)^2 \right. \right.$$

$$\left. + \theta \right\} \qquad \text{(AI-16)}$$

$$\left(\frac{z_D}{R}\right)_V \approx \left\{ \frac{dZ_A}{dX_A} \left[1 - 0.45 \left(\frac{dY_A}{dX_A}\right)^2 \right] + \theta \right\} - \phi \left\{ \frac{dY_A}{dX_A} \left[1 - \frac{1}{4} \left(\frac{dY_A}{dX_A}\right)^2 \right] \right.$$

$$\left. - \psi \right\} \qquad \text{(AI-17)}$$

Equations AI-16 and AI-17 are recommended if the ground track is not to make an angle of more than 40 deg with the extension of the runway center. For full freedom of direction of flight Equations AI-13, AI-14, and AI-15 might have to be used. This is an area where further study might be expended.

The equation and the approximate equation for the horizon line follow. The approximate equation is recommended

$$\left(\frac{z_D}{R}\right)_H = \frac{\sin\theta}{\cos\phi} - \left(\frac{y_D}{R}\right)_H \tan\phi \qquad \longleftarrow \text{exact horizon}$$

$$\approx \theta - \left(\frac{y_D}{R}\right)_H \phi \tag{AI-18}$$

The equations for the altitude spot are

$$\left(\frac{z_D}{R}\right)_A = Kh \quad h < h_o \; ; \left(\frac{z_D}{R}\right)_A = \pm 1 \quad h > h_o \tag{AI-19}$$

$$\left(\frac{y_D}{R}\right)_A = \left(\frac{y_D}{R}\right)_V \tag{AI-20}$$

2. INFORMATION SET

This set of equations is given for information only. The radius of the velocity circle is ρ, which is a constant. X is the component of the horizontal range to the center of the far end of the runway along the projection of the aircraft x body axis. Y is the horizontal component perpendicular to X, defined positive so that X, Y, and Z constitute a right-hand coordinate system and Z is positive up.

The reference display is shown in Figure AII-19.

Horizon

$$y_d = x_d \tan \phi - K_\theta \theta \tag{AI-21}$$

Velocity circle

$$\rho^2 = \left\{ x_d - K_\theta \left[\cos\phi \tan^{-1}\frac{\dot{Y}_h}{\dot{X}_h} - \sin\phi \tan^{-1}\frac{\dot{h}}{\dot{X}_h} \right] \right\}^2$$

$$+ \left\{ y_d - K_\theta \left[\sin\phi \tan^{-1}\frac{\dot{Y}_h}{\dot{X}_h} - \cos\phi \tan^{-1}\frac{\dot{h}}{\dot{X}_h} - \theta \right] \right\}^2$$

Runway

Takeoff end

$$y_d = x_d \tan\phi - K_\theta \left[\theta + \frac{\tan^{-1}\frac{h}{X}}{\cos\phi} \right]$$

Intermediate line

$$y_d = x_d \tan\phi - K_\theta \left[\theta + \frac{\tan^{-1}\left(\frac{h}{X - a\cos(\psi_o - \psi)}\right)}{\cos\phi} \right]$$

Landing threshold

$$y_d = x_d \tan\phi - K_\theta \left[\theta + \frac{\tan^{-1}\left(\frac{h}{X - \ell\cos(\psi_o - \psi)}\right)}{\cos\phi} \right]$$

$$A = K_\theta \left[\tan^{-1}\frac{Y}{X}\cos\phi + \tan^{-1}\frac{h}{X}\sin\phi \right]$$

$$B = K_\theta \cos\phi \frac{b\cos(\psi_o - \psi - \zeta_h)}{R_h}$$

$$C = K_\theta \left[\tan^{-1}\frac{Y - a\sin(\psi_o - \psi)}{X - a\cos(\psi_o - \psi)}\cos\phi + \tan^{-1}\frac{h}{X - a\cos(\psi_o - \psi)}\sin\phi \right]$$

$$D = K_\theta \cos\phi \frac{b\cos(\psi - \psi - \zeta_h - \tau)}{R_h'}, \quad \tau = \sin a \frac{\sin(\psi_o - \psi - \zeta_h)}{R_h'}$$

$$E = K_\theta \left[\tan^{-1}\frac{Y - \ell\sin(\psi_o - \psi)}{X - \ell\cos(\psi_o - \psi)}\cos\phi + \tan^{-1}\frac{h}{X - \ell\cos(\psi_o - \psi)}\sin\phi \right]$$

$$F = K_\theta \cos\phi \frac{b\cos(\psi_o - \psi - \zeta_h - \omega)}{R_h''}, \quad \omega = \sin^{-1}\ell \frac{\sin(\psi_o - \psi - \zeta_h)}{R_h''}$$

Associated and supporting equations

$$\dot{X}_h = \dot{X} \ , \ \dot{Y}_h = \dot{Y} \qquad \zeta_h = \tan^{-1} \frac{\sin \zeta \ \cos \phi}{\cos \theta \sqrt{1 - \sin^2 \zeta \cos 2 \phi}}$$

$$R_h = \sqrt{X^2 + Y^2} \qquad R_h' = \sqrt{a^2 + R_h^2 - 2a\, R_h \cos(\psi_o - \psi - \zeta_h)}$$

$$R_h'' = \sqrt{\ell^2 + R_h^2 - 2\ell \ R_h \cos(\psi_o - \psi - \zeta_h)}$$

Making a small angle approximation results in the following equations:

Horizon

$$y_d = \phi \ x_d - K_\theta \theta$$

Velocity circle

$$\rho^2 = \left(x_d - K_\theta \ \frac{\dot{y}}{V}\right)^2 + \left(y_d - K_\theta \left[\frac{\dot{h} + \phi \dot{y}}{V} - \theta\right]\right)^2$$

Runway

Takeoff end

$$y_d = \phi \ x_d - K_\theta \left(\theta + \frac{h}{X}\right)$$

Intermediate line

$$y_d = \phi \ x_d - K_\theta \left(\theta + \frac{h}{X - a}\right)$$

Landing threshold

$$y_d = \phi \ x_d - K_\theta \left(\theta + \frac{h}{X - \ell}\right)$$

$$A = K_\theta \left(\frac{Y}{X} + \frac{h}{X} \phi\right)$$

$$B = K_\theta \ \frac{b}{R_h''}$$

$$C = K_{\theta} \left(\frac{Y - a\,(\psi_o - \psi) + h\,\phi}{X - a} \right)$$

$$D = K_{\theta} \frac{b}{R_h{}'}$$

$$E = K_{\theta} \left(\frac{Y - \ell\,(\psi_o - \psi) + h\,\phi}{X - \ell} \right)$$

$$F = K_{\theta} \frac{b}{R_h{}''}$$

$$R_h = \sqrt{X^2 + Y^2} \qquad R_h{}' = R_h - a \qquad R_h{}'' = R_h - \ell$$

APPENDIX II. ANALOG SIMULATION

An analog simulation study was conducted primarily to investigate the effectiveness of the manual blind landing system situation display. The techniques and procedures used for this investigation are briefly described below.

A. PERFORMANCE OF ANALOG SIMULATION

The services of "pilots" in the following three categories were obtained: (1) nonpilots familiar with the pilot's task, (2) part-time pilots, and (3) full-time pilots (all subjects who "flew" the simulator are referred to as pilots). A standard routine was established for each pilot to allow comparison of their performance. Statistical methods were used to reflect the success of the landings through compilations of aircraft attitudes, position and rates at touchdown.

The equations presented later in this appendix were mechanized on the Autonetics Simulation Facility (shown in Figure AII-1). The Blind Landing System simulator consisted of a small screened-off cockpit (Figure AII-2), three Electronic Associates analog computers (Type 16-31R), two 6-channel Offner Electronics, Inc., recorders (Type MC).

Figure AII-1. Autonetics Simulation Facility

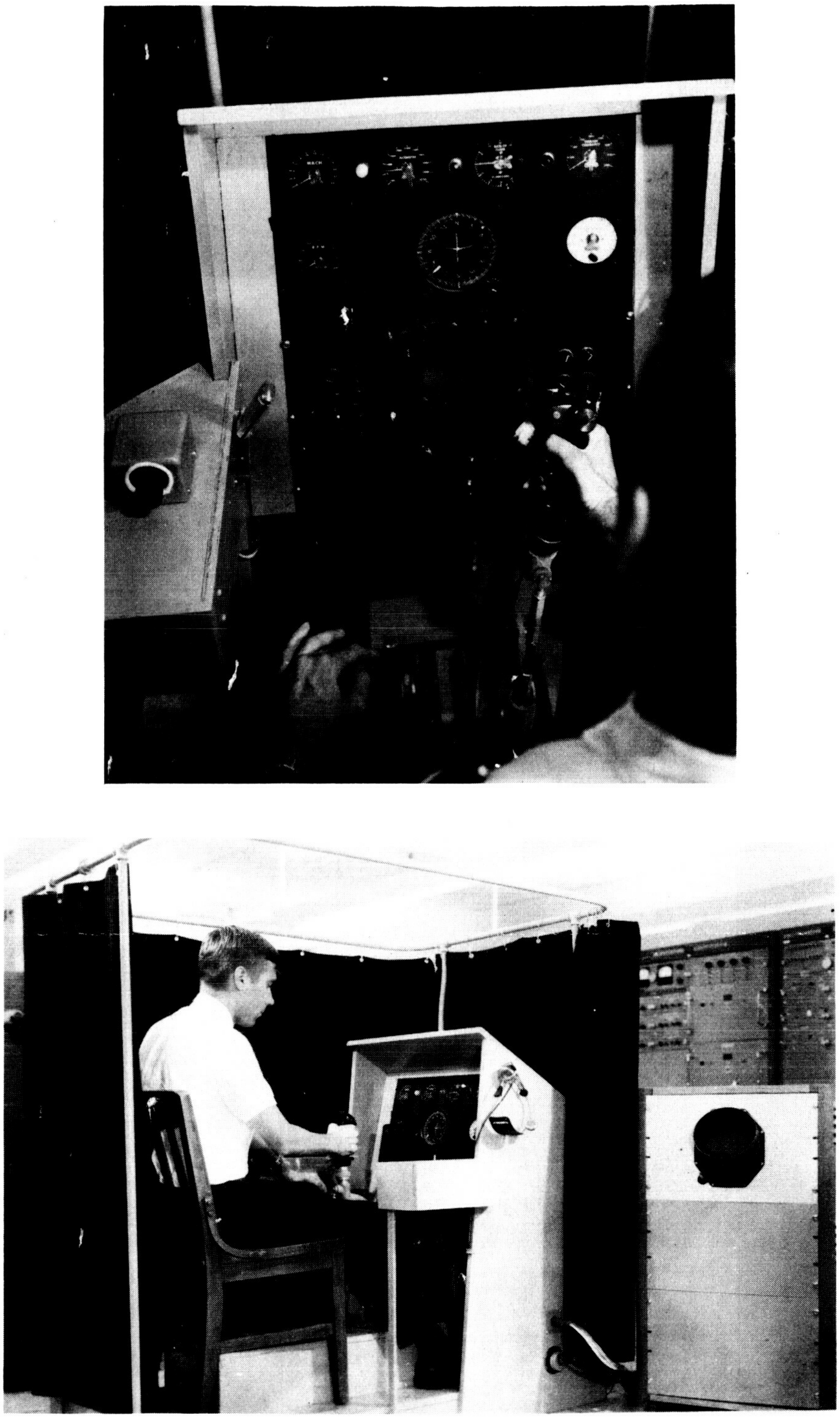

Figure AII-2. Blind Landing System Simulator - Cockpit Displays two Mosley x-y plotters, a paralleled situation display for the computer engineer to observe, and various servoamplifiers and resolvers. The analog mechanization is given in Figure AII-3.

Pilots flying the simulator were given essentially the following information: This is a manual blind landing system simulation and as such it tries to recreate what is seen through the windshield of the airplane under clear weather conditions. The heart of the system is the display (see Figure AII-4). The display contains information that would be seen through a similar sized aperture in the windshield if the windshield were painted black except for a circle the size of the display.

The display contains an artificial horizon. This is similar to the artificial horizon in a real airplane. If the simulated aircraft is banked right wing low, the horizon rotates counterclockwise. Similarly, if the nose of the aircraft is pulled up, the horizon moves down.

The display also contains an airplane symbol. This is fixed to the center of the display. The vertical distance between the horizon line and the airplane symbol is the pitch angle. The rotation of the horizon line with respect to the airplane symbol is the bank angle. The sense of these two angles is interpreted as explained before..

The small circle on the display is called the velocity circle. It shows the direction in which the aircraft's inertial velocity is pointing with respect to the aircraft. The vertical displacement of this circle with respect to the aircraft symbol is the angle of attack, while the vertical displacement with respect to the horizon is the flight path angle. The lateral displacement of the velocity circle reflects the effects of side slip angle, crab angle, and crosswinds. When the velocity circle is below the horizon, it represents the point where the aircraft would hit the ground if it were left undisturbed. This will be useful in maintaining a glide path.

The runway is represented by four lines. The runway display corresponds conceptually to the view of the runway at night if it had four neon tubes on it. One of these ficticious neon tubes would be down the runway centerline. The other three neon tubes can be thought of as being placed across the runway; one at the threshold, one at the middle and one at the far end. The relative motion between the aircraft and the runway is seen as apparent motion of the runway just as it would when flying under visual flying rules (VFR). Thus, in lining up with the runway if the runway appears off to the right, one flies to the right.

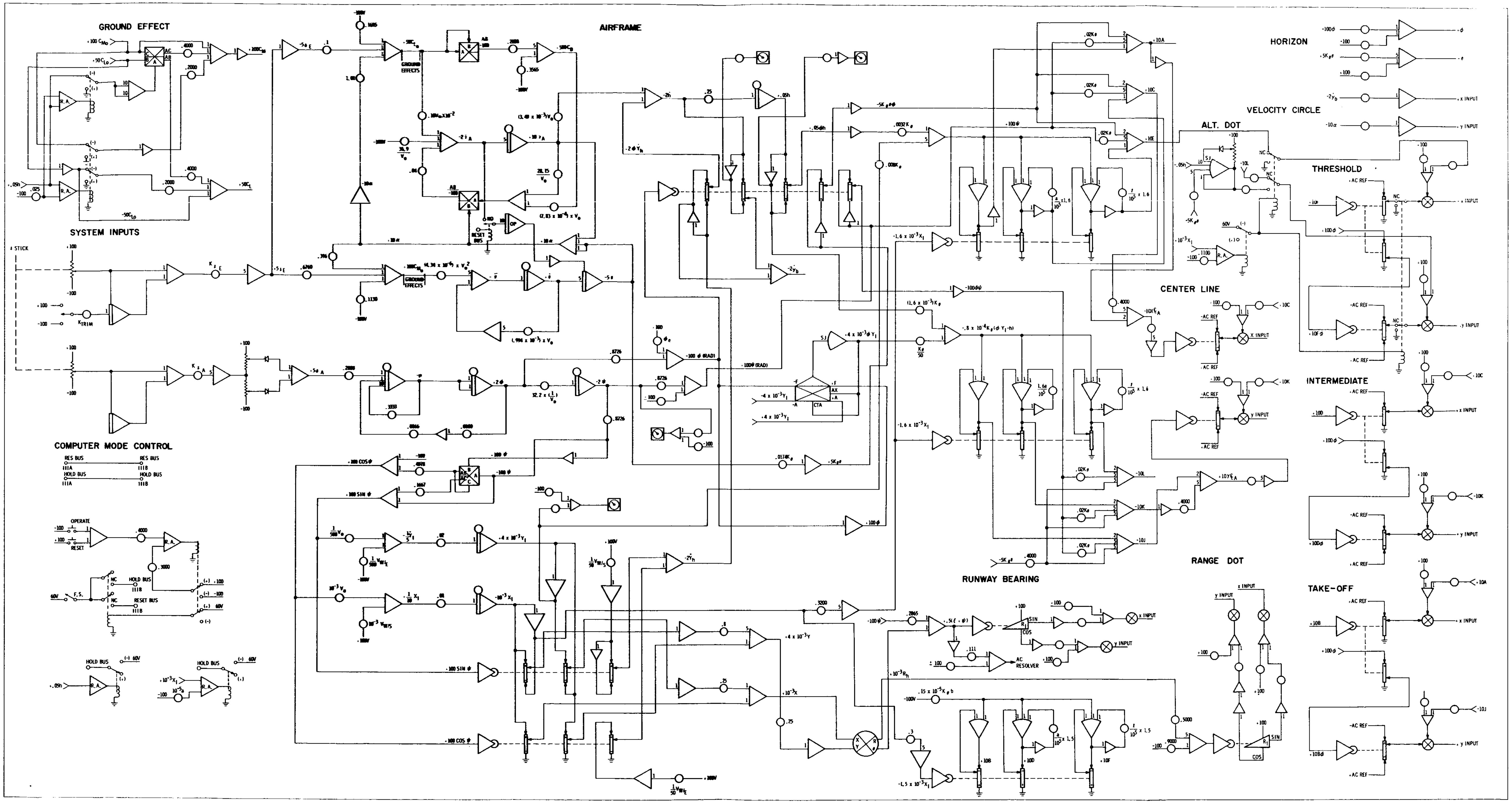

Figure AII-3. Blind Landing System - Analog Mechanization

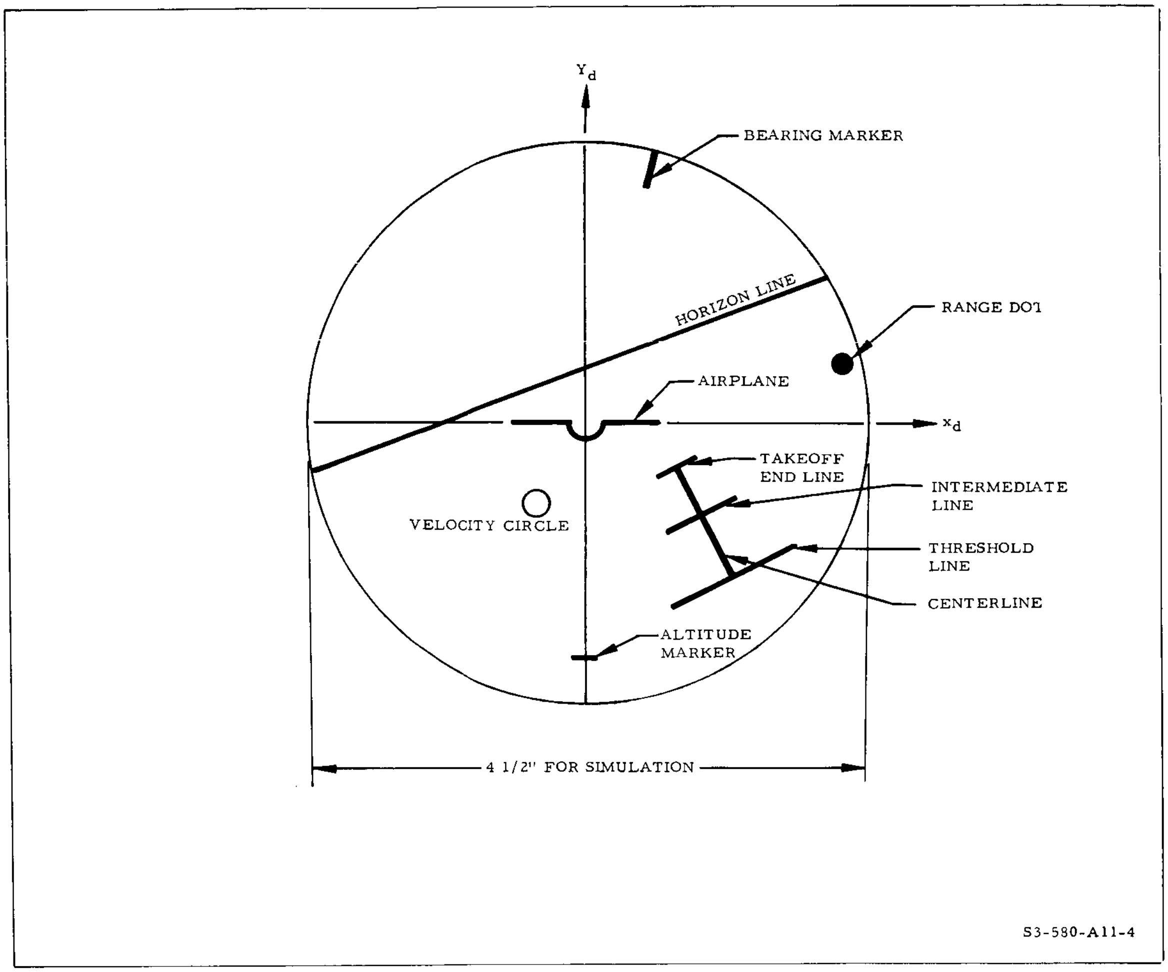

Figure AII-4. CRT Landing Display

There is an indication of range from the far end of the runway on the display. This is a spot that appears around the periphery of the scope. This spot is calibrated such that when the spot is in the 12 o'clock position the aircraft is at the far end of the runway. The 3 o'clock position corresponds to a range of 3 naut mi from the far end of the runway, the 6 o'clock position to 6 naut mi from the far end of the runway, etc. There is a scribed mark on the plexiglass tube cover which indicates the runway length so that distance from the runway threshold can also be seen.

On the periphery of the display, there is also a radial line segment. This radial line segment is an indication of the aircraft's bearing from the far end of the runway relative to the runway. This bearing angle is the horizontal projection of the angle formed by the runway centerline and a line from the center of the far end of the runway to the aircraft and is analogous to a conventional localizer error but is not limited by localizer beam width.

The last symbol on the display is an altitude spot. This spot appears on the display when the aircraft is 1,000 ft from the runway threshold. The altitude spot is referenced to the horizon and full scale deflection is nominally 100 ft.

The first step in the approach and landing maneuver is lining up the aircraft with the runway centerline. If the runway lines are off the display, the runway would be outside the field of vision afforded by a similar sized window. Line-up can still be accomplished by using the bearing marker. If the bearing marker is to the right, the runway is to the right. Thus, in order to line up, the aircraft is flown to the right until the runway is in view. Then a normal VFR type approach can be flown.

The following method may be used to establish the proper glide path. An easily identifiable point in space on the desired glide path is chosen. The aircraft is then flown at the proper altitude, nominally about 1300 ft. This altitude may be maintained by keeping the velocity circle on the horizon. At the proper range, nominally about 6 naut mi from the far end of the runway, the desired sink rate is established. If the runway is on the display, the sink rate is established by pushing the controls forward until the velocity circle is on the runway threshold line. If the runway is not on the display letdown is set up so that the desired sink rate, nominally about 600 ft/min is achieved (sink rate being read off the appropriate instrument). By keeping the velocity circle the same distance below the horizon when the runway is off the display or at the same vertical displacement as the runway threshold line when the runway is on the display, the aircraft can be flown down the desired glide path.

As the runway is approached it appears to grow larger. This tends to increase the sensitivity of the display. The runway centerline should be kept vertical and in line with the velocity circle. If the runway centerline makes a positive angle with the vertical, positive being defined as a required clockwise rotation to go from the vertical to the runway centerline, this indicates that the runway is to the left. The opposite is also true.

When the altitude spot starts to move, the aircraft is at about 100 ft of altitude. The range spot is observed to verify that the aircraft will not be short of the runway. A flare is performed by allowing the altitude spot to get slightly above the velocity circle. Control action is then initiated so that the distance between the two symbols is maintained at between 1/16 in. and 1/8 in. This will cause a sink rate at touchdown of between 1 and 3 ft/sec.

Statistical data on the pilot's performance using the display was gathered as follows. At least 1/2 hr simulator time was granted prior to flying a standardized condition. Ten trial flights with the standard initial condition were then performed before recording two sets of production runs (each set, ten runs) from which statistical data were obtained.

The standard initial condition provides reasonable circumstances and assumes that the pilot has navigated to a position from which display information will be easily available. Parameters for this initial condition are as follows:

X_I = 36700 ft

Y_I = 1500 ft

h = 1000 ft

ψ = 9.3 deg

θ = ∞ (i.e., trimmed for level flight)

ϕ = 0

Typical plots of aircraft position from the standard initial conditions to touchdown are given in Figures AII-5 and AII-6. Recorded traces of the variables for the same conditions are presented in Figure AII-7.

Initial tendency of nonpilots to inadvertently apply reverse stick commands diminished rapidly with practice. Glide slopes could easily be set up and maintained using either range, altitude, and sink rate information or velocity circle and runway threshold line.

Safe landings were possible with initial conditions which required violent maneuvers immediately prior to touchdown. These initial conditions are as follows:

X_I = 15000 ft

Y_I = 500 ft

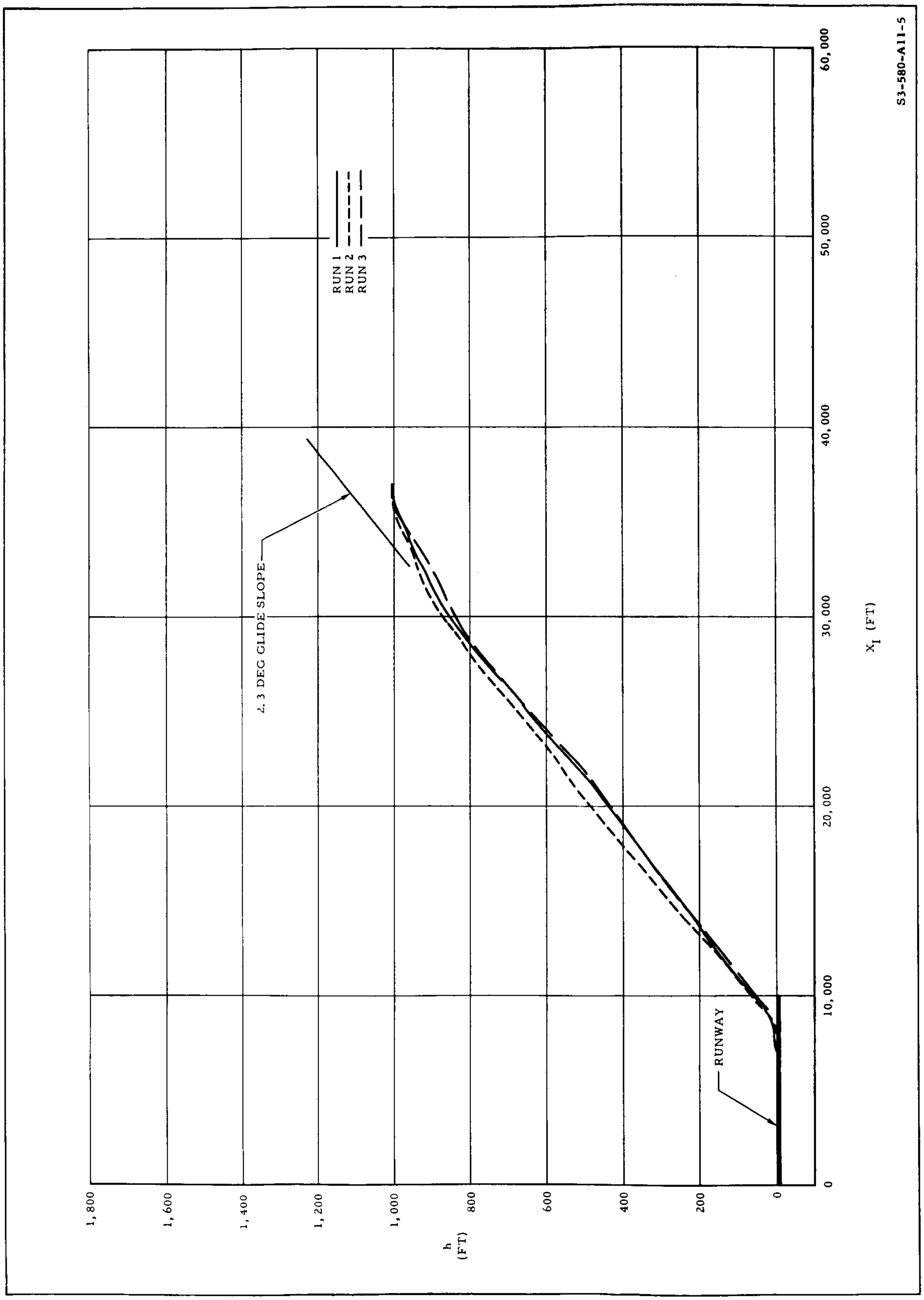

Figure AII-5. Plots of h vs X_i for Standard Initial Condition

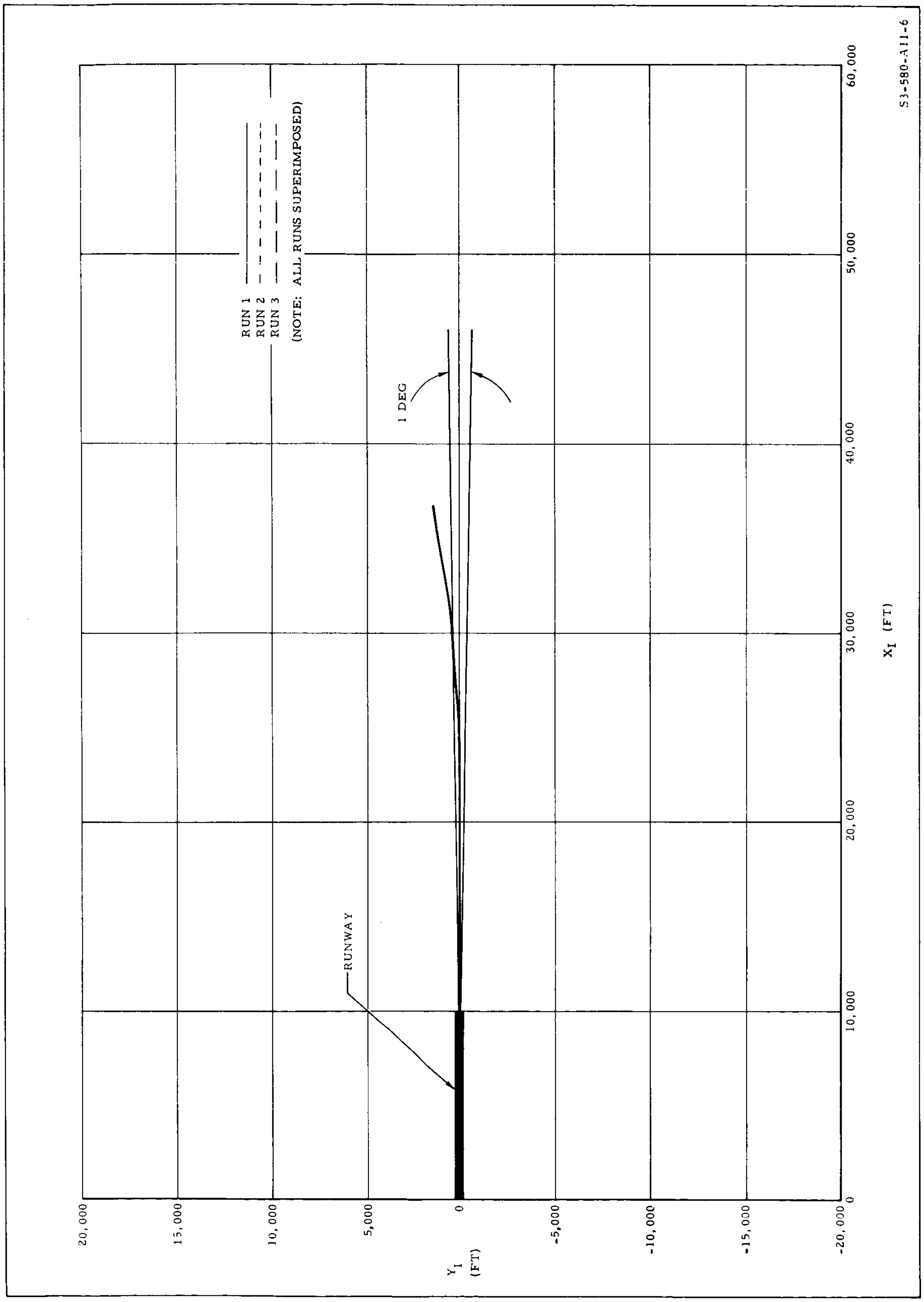

Figure AII-6. Plots of Y_i vs X_i for Standard Initial Condition

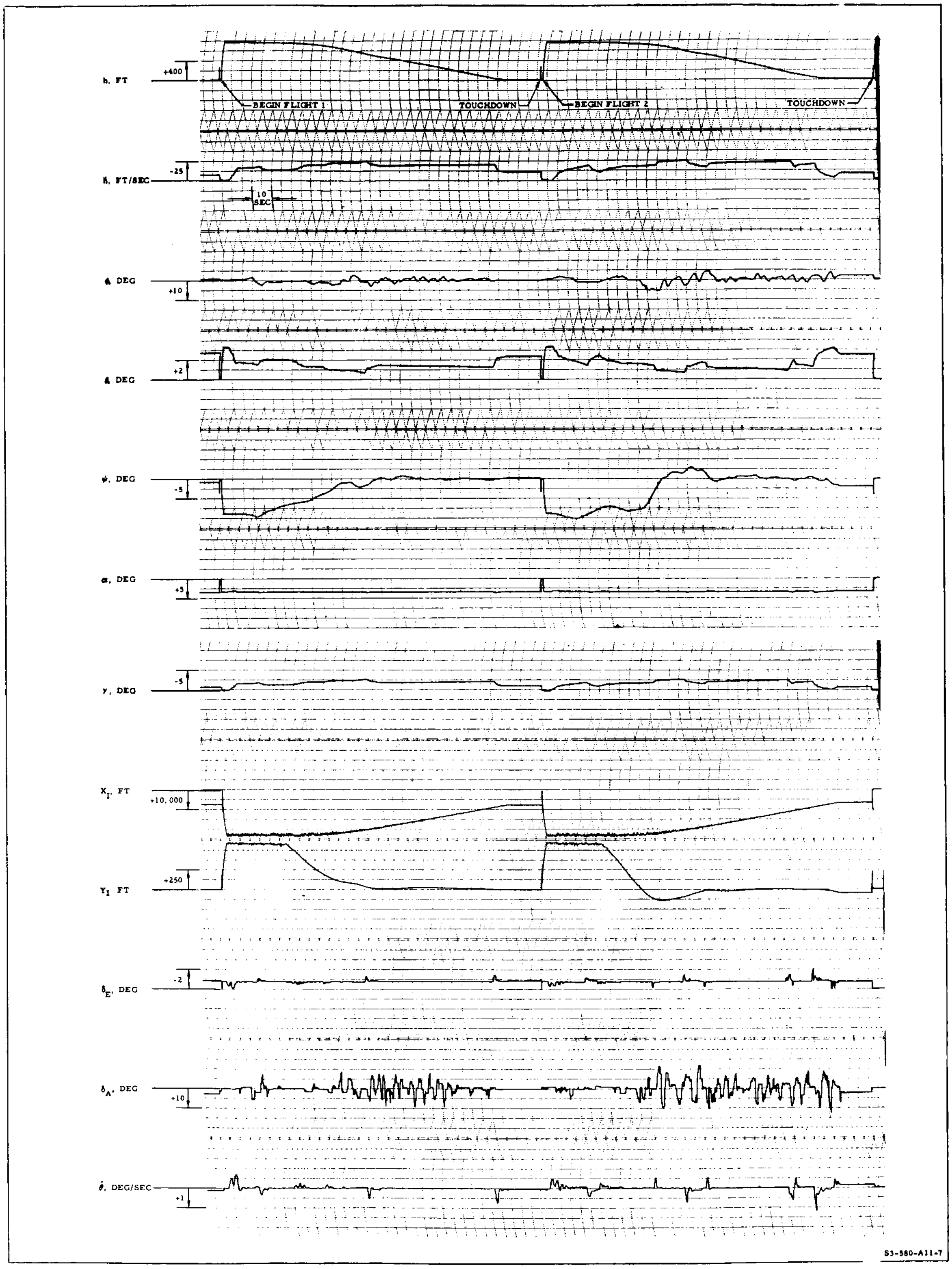

Figure AII-7. Time History of Variables for Two Successive Flights

$h = 660$ ft

$\psi = 0$

$\theta = \infty$

$\phi = 0$

Safe landings were also possible from initial conditions which required navigational tasks prior to getting the runway lines on-scope. Plots of aircraft position are given in Figures AII-8 and AII-9. The initial conditions are as follows:

$X_I = 50,400$ ft

$Y_I = 17,600$ ft

$h = 1,000$ ft

$\psi = 9.3$ deg

$\theta = \infty$

$\phi = 0$

Safe landings were also recorded for an STOL configuration. The initial conditions are:

$X_I = 17,000$ ft

$Y_I = 0$

$h = 1,900$ ft

$\theta = \infty$

$\phi = 0$

$\psi = 0$

Aircraft forward velocity was maintained at 200 ft/sec for all landings. Martin 404 landing speed is reported as being 116 ft/sec; Convair CV-240 is 129 ft/sec. At 200 ft/sec, the velocity imposes rather strict demands on the pilot but is nearly the same as for current near-sonic transports. To assess the dispaly effectiveness for larger aircraft, static gains and vehicle dynamics were altered arbitrarily to a configuration that was more sluggish in roll. No statistical data was taken, however, the display appeared to be nearly as effective under these conditions as for the conditions reported in succeeding pages.

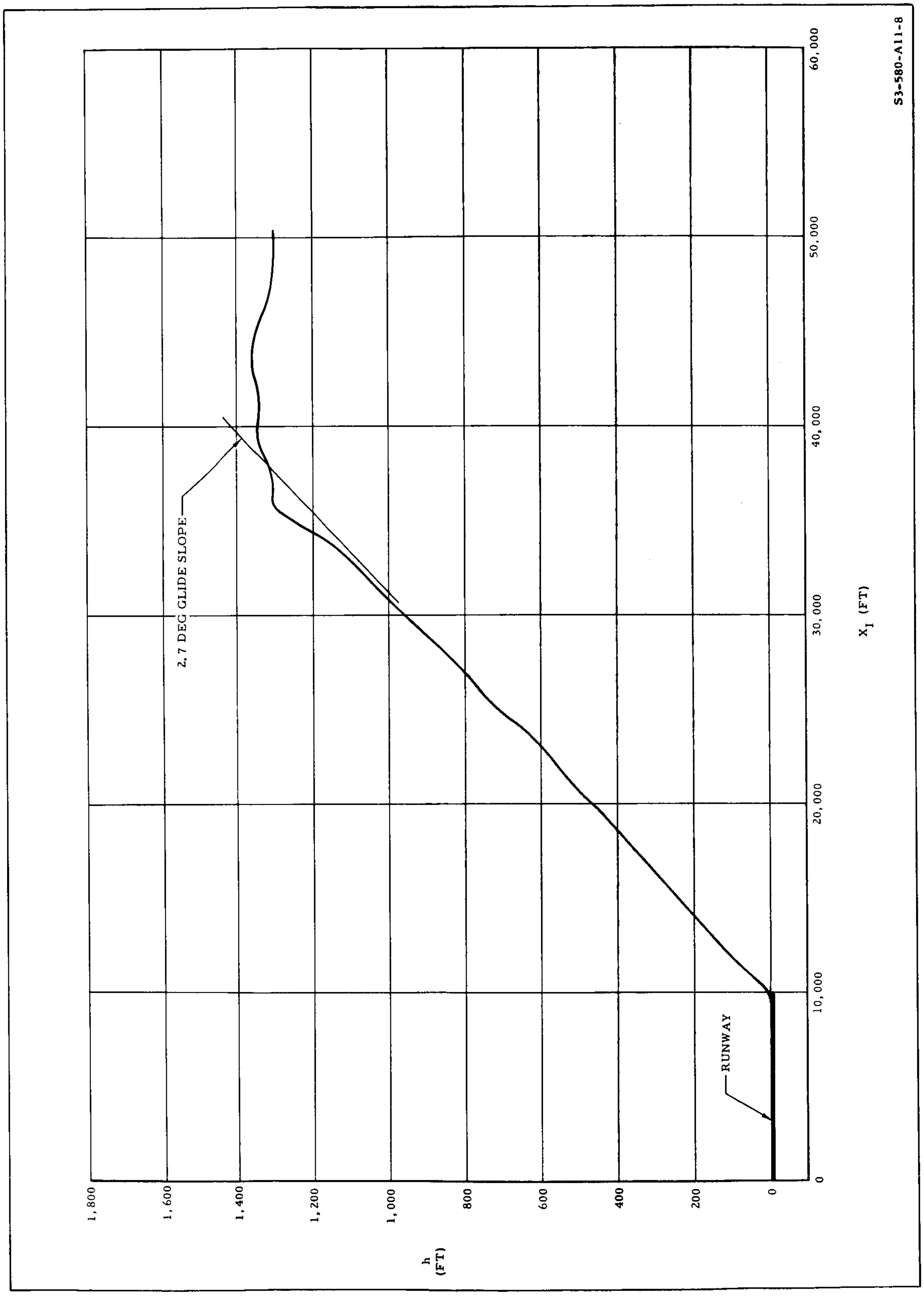

Figure AII-8. **Plot of h vs X_I for Navigational Initial Conditions**

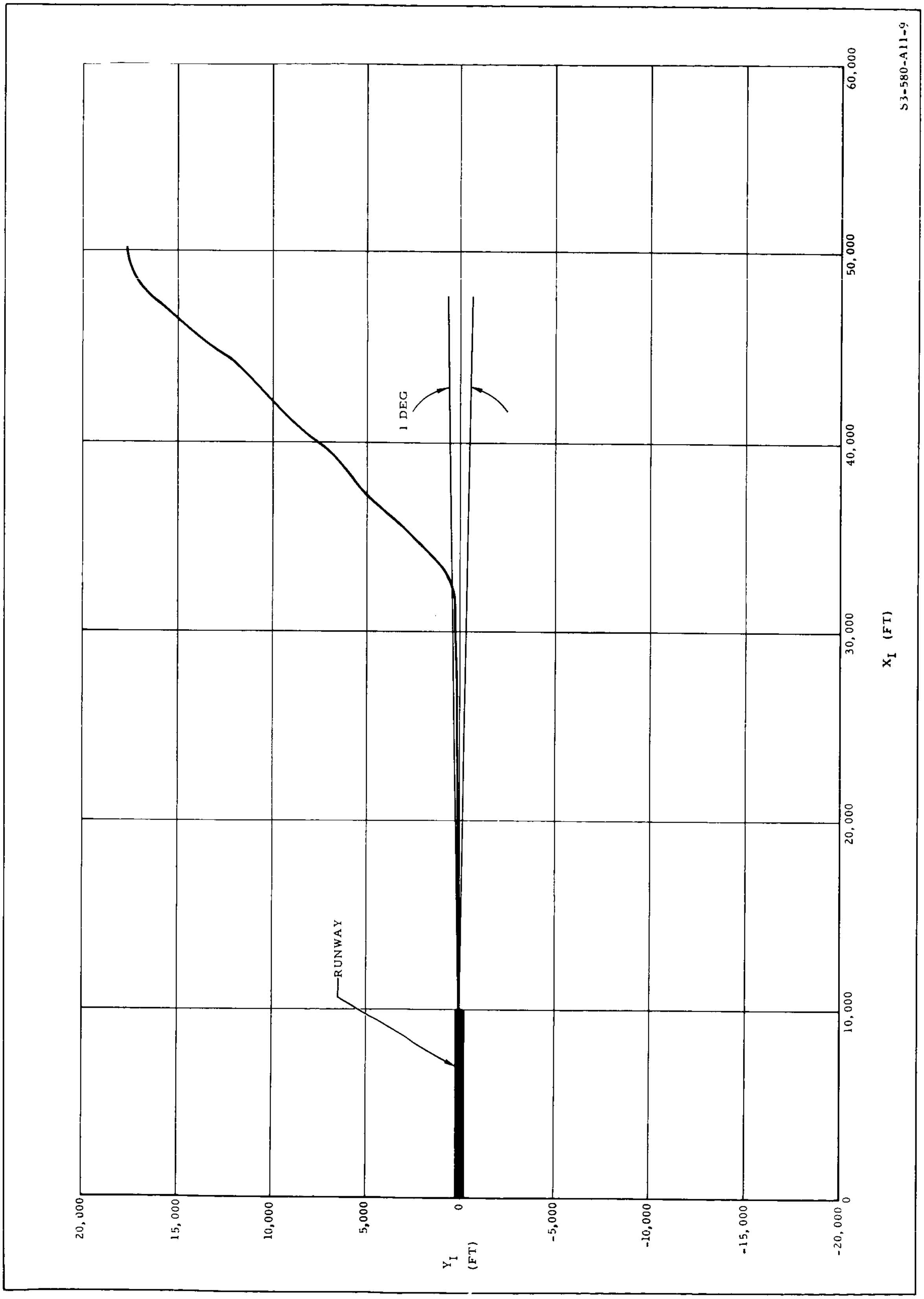

Figure AII-9. Plot of Y_i vs X_i for Navigational Initial Conditions

A range marker, bearing marker, and runway centerline, all shown in Figure AII-4, were added to the original display in the course of the simulation. These additions were found to be useful in the following ways. The range marker gave the pilot an idea of the degree of maneuvering necessary to line up with the runway, aided in setting up an accurate glide path, and helped the pilot to tell if he was in danger of landing short of the runway. The bearing marker was helpful in maneuvering to line up with the runway when the runway was off the display. A meter was also installed in the cockpit which had lateral displacement from the runway and was helpful in this task. The information need to drive the meter is available and the pilot might find it useful for other tasks. The runway centerline gives a much higher apparent sensitivity to the display when near the runway. It is felt that it is a useful inclusion to the display.

Flights were made which included moderate gusts and wind components from South and East. The display presents these conditions in the form of a laterally offset and fluctuating velocity circle. It was not difficult to correct for moderate gusts and wind. However, because of lack of rudder control, decrab maneuvers were not possible so flights resulted in touchdowns with heading error proportional to lateral wind velocity.

The following sensor errors were injected into the display equations for aircraft angular terms. Several flights were made for each error.

$\Delta \phi$ = 1 deg

$\Delta \theta$ = 1 deg

$\Delta \psi$ = 2 deg

The roll error resulted in horizon and runway lines rotated 1 deg from nominal. This condition presented no serious problem to the pilot. The roll angle causes yaw or heading error initially, but the pilot readily adapts to the condition by relying more on the velocity circle intersection with ground points, and actually flying level, with the horizon and runway lines rotated slightly, to avoid lateral drift of the velocity circle. This corrective action is evidently maintained through to touchdown, bacause touchdown roll angle did not reflect the injected error.

The pitch error resulted in horizon and runway lines displaced vertically 1 deg (approximately 1/4 in.) from nominal. The velocity circle does not contain this error, therefore, steering with it is erroneous. Touchdowns were adequately executed for all of the simulator runs. For an aircraft installation, angle of attack (which drives the velocity circle vertically) is a function derived from pitch angle, so that any display errors resulting from incorrect pitch angle will be in all major references including altitude marker on the scope face. This means there will be no relative error, so landing performance will not be jeopardized.

The yaw error resulted in erroneous heading of runway lines which was maintained through to touchdown and resulted in an average touchdown error of 2 deg heading.

B. EVALUATION OF DATA

The ultimate performance of the pilots is not indicated by the simulation runs. Figure AII-10 represents a learning curve of the group as a whole with the implication that percentage of unsuccessful landings is an inverse measure of proficiency development. Go arounds were not considered unsuccessful landings. It is noted that the curve is not yet flat, indicating that a longer learning period is required.

A desirable display design is one for which operator learning requires the shortest time. For the Ames display, the criterion is apparently met as evidenced by the sharp slope of the dashed line approximation on Figure AII-10. The dashed line indicates proficiency improvement of at least 30 percent relative to each previous set of four runs.

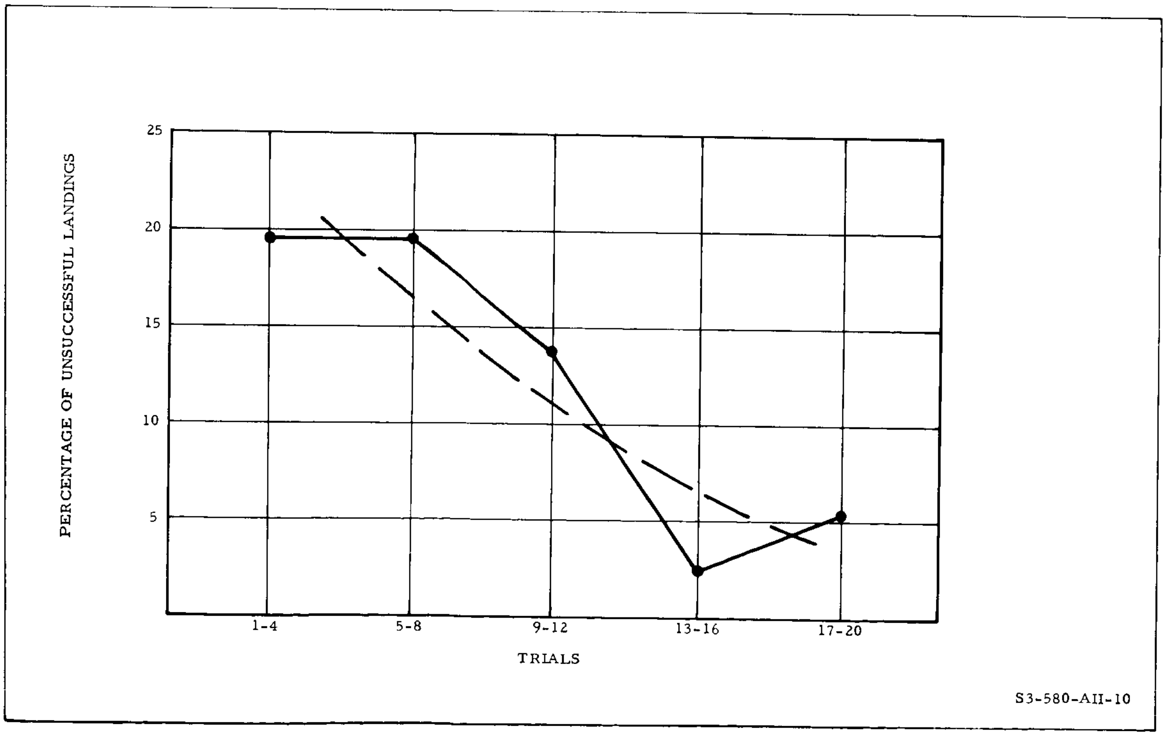

Figure AII-10. Learning Curve Based on Percentage of Unsuccessful Landings

Table AII-1 presenta a tabulation of percentage of safe touchdowns, percentage of go-arounds, average number of times roll angle exceeded ±5 deg, and average number of times roll angle crossed through zero. These may be interpreted as follows:

1. Number of attempts. This represents the number of production runs initiated using the standard initial condition.

2. Number of touchdowns. This is equal to the number of attempts less the number of go-around flights. No touchdowns were made off the runway.

3. Percentage of touchdowns which were safe. This indicates the number of landings for which touchdown parameters fell within the following ranges:

$$5000 \leq X_I \leq 10000 \text{ ft}$$

$$|Y_I| \leq 75 \text{ ft}$$

$$\dot{h} \leq 8 \text{ ft/sec}$$

$$|\theta| \leq 2.5 \text{ deg}$$

$$0 \leq \theta \leq 5 \text{ deg}$$

$$|\psi| \leq 2.5 \text{ deg}$$

   It is seen that the more experienced pilots had 95 percent safe touchdowns. Because of the learning process associated with flying to the display, it is reasonable to assume that this measure of success will approach 100 percent after additional simulator training. If a greater number of runs had been made, the value for part-time pilots (unexpectedly low) may have more appropriately fallen between those for full-time pilots and non-pilots.

4. Percentage of flights which resulted in go-around. These figures appear to indicate that the two full-time pilots are much more selective of touchdown conditions than are part-time and non-pilots. Validity of this assumption is reflected in values of touchdown parameters in Table AII-1.

Table AII-1. Tabulation of Safe Touchdowns, Go-Around, and Roll Activity

	Number of Flights	Percentage of Flights Which Resulted in Go-Arounds	Average Number of ϕ -5 Deg Exceedances Per Flight	Average Number of ϕ = Zero Crossings Per Flight	Number of Touchdowns	Percentage of Touchdowns Which Were Safe
Overall	198	7.57	1.44	22.7	183	85.8
Full-Time Pilots (2)	46	13.04	3.3	28.8	40	95.0
Part-Time Pilots (3)	66	6.06	1.2	26.3	62	80.7
Nonpilots (4)	86	5.8	0.6	16.8	81	85.1

5. Average number of $|\phi| = 5$ deg exceedances per flight, and (δ) average number of θ = zero crossing per flight. These conditions were conceived as further measures of pilot proficiency, however, it is seen that they both exhibit larger values for the more experienced pilots. Interpretation of these results is difficult. It may be said that the experienced pilot is more apt to enter small corrective commands that require a greater number of zero crossings.

Table AII-2 presents mean and standard deviation of variables at touchdown. It is based on production runs using the standard initial condition. Used in conjunction with the histograms of Figures AII-11 through AII-16 (based on the same data) considerable insight into display effectiveness is provided. The histograms indicate that even with limited training, reasonable landings may be expected.

The histogram for ψ (yaw angle), shows it to be the only parameter whose distribution approaches being Gaussian. It is expected that the other variables would also have a Gaussian distribution for a larger sample. The standard deviation, or "one-sigma" values reported in Table AII-2 apply only for such a distribution and are, therefore, only trend indicators for each variable.

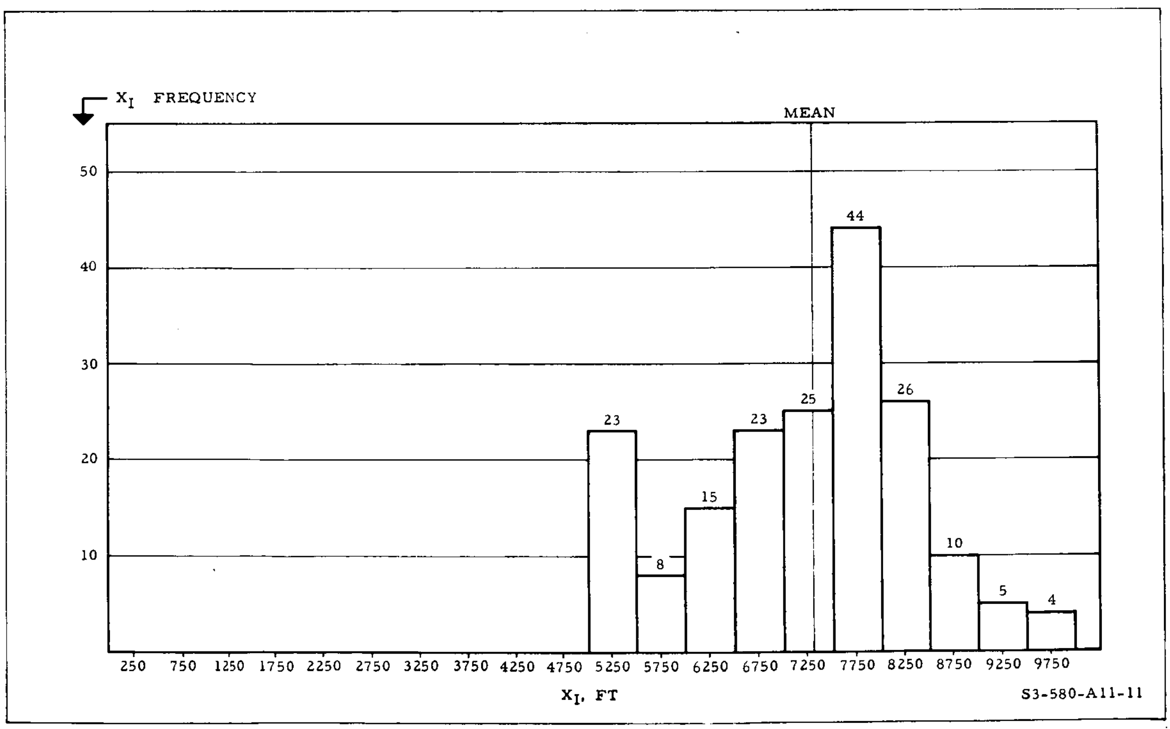

Figure AII-11. Histogram of X_i at Touchdown

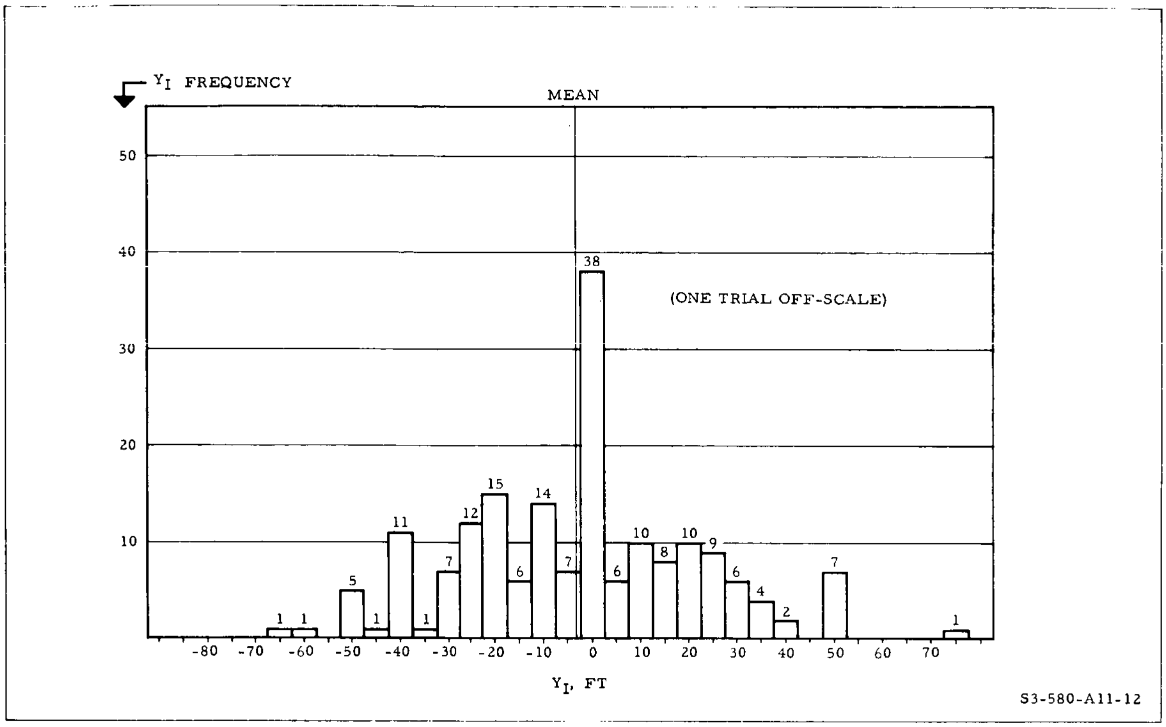

Figure AII-12. Histogram of Y_i at Touchdown

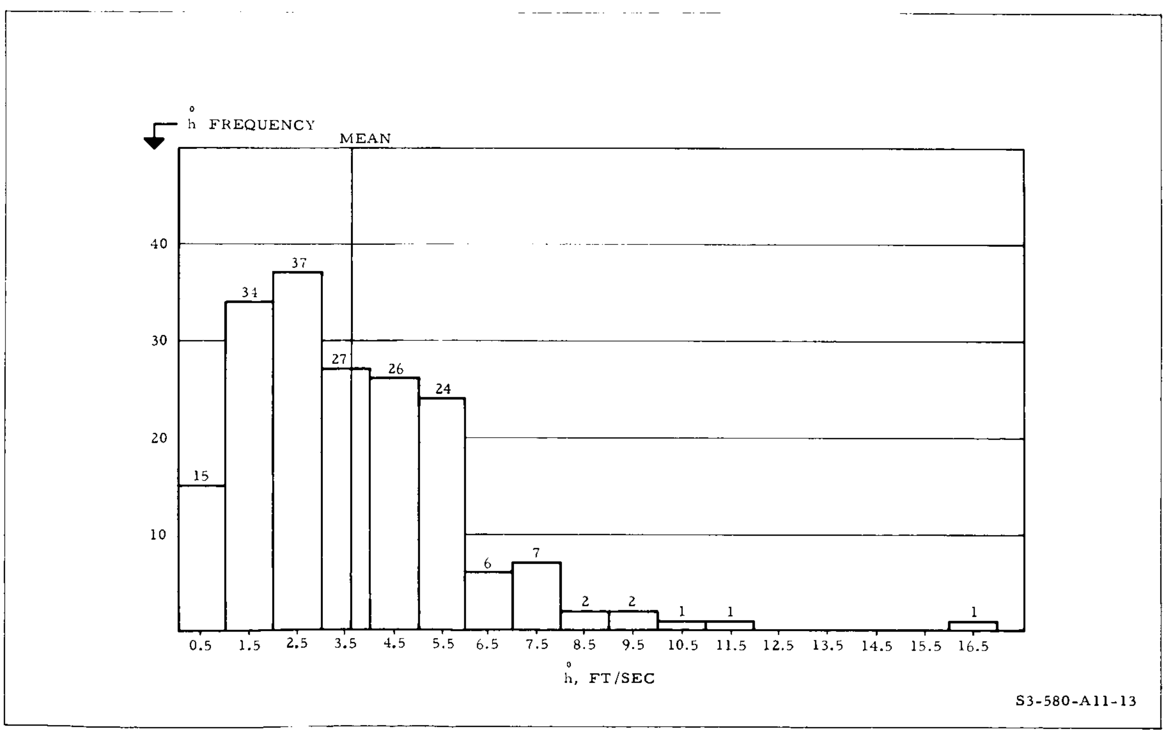

Figure AII-13. Histogram of $\dot{h}$ at Touchdown

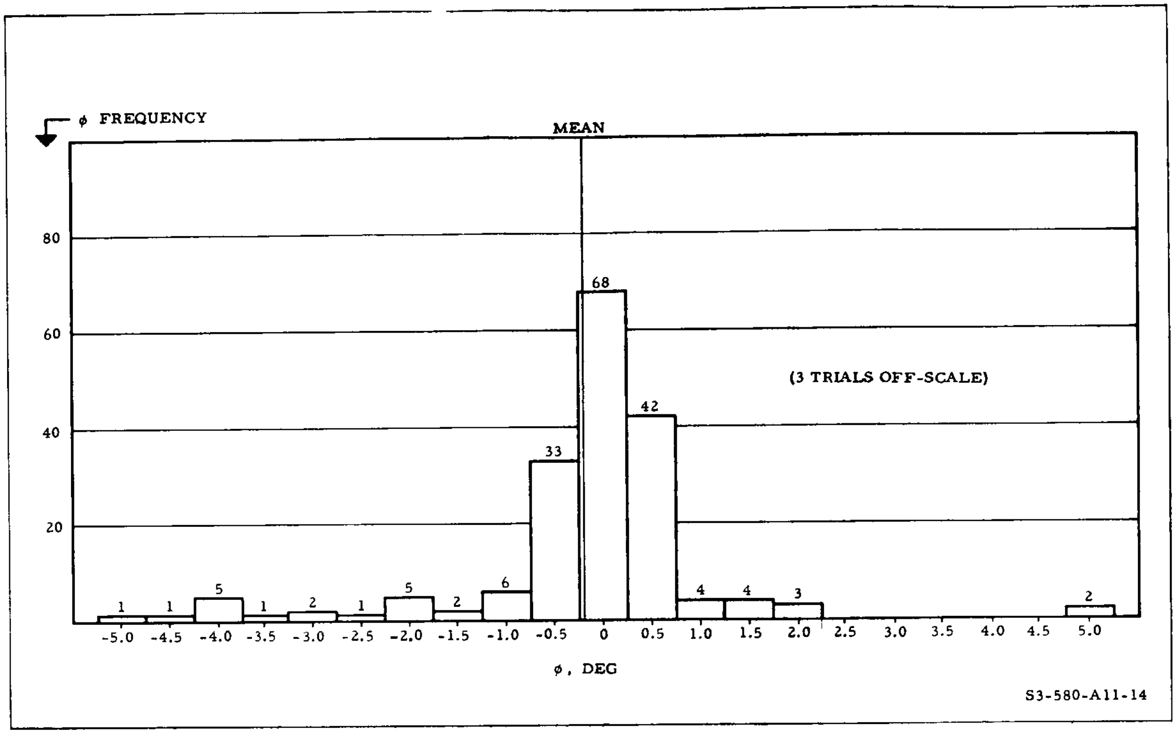

Figure AII-14. Histogram of ϕ at Touchdown

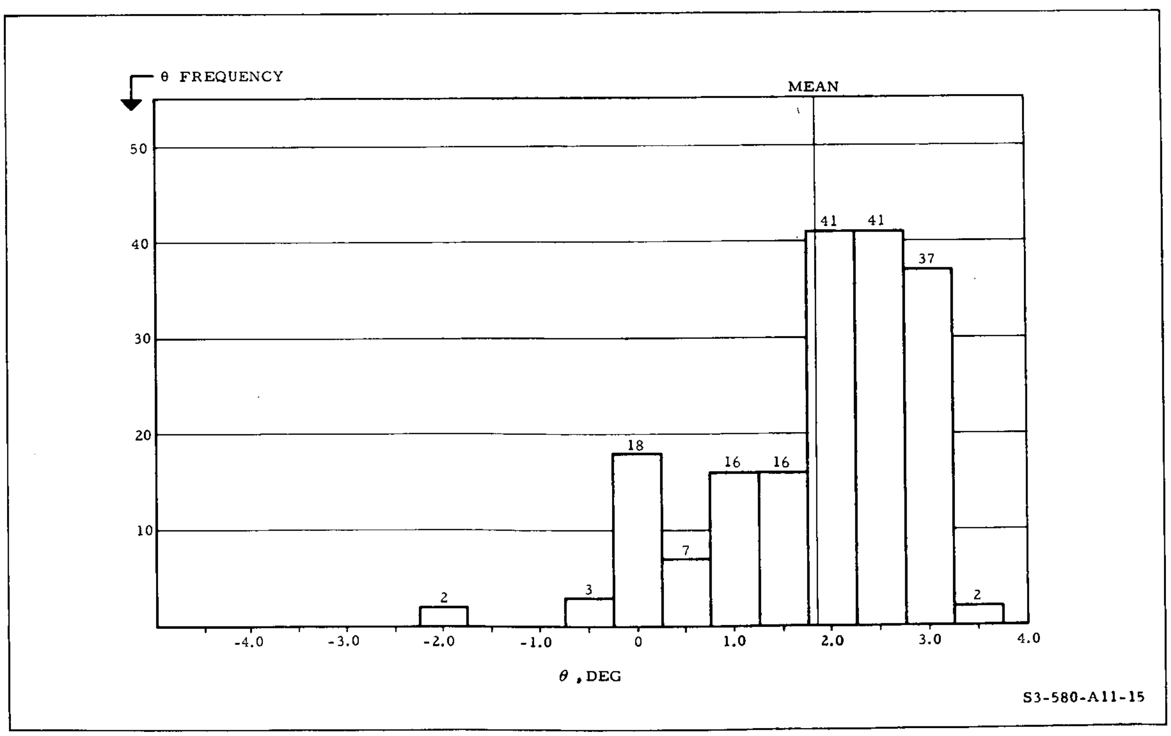

Figure AII-15. Histogram of θ at Touchdown

Table AII-2. Mean Values and Standard Deviation of Variables at Touchdown

	X_I Mean	Standard Deviation	Y_I Mean	Standard Deviation	h^o Mean	Standard Deviation	Mean	Standard Deviation	ϕ Mean	Standard Deviation	Mean	Standard Deviation
Overall	7330	1148	-3.36	26.2	3.94	2.38	-0.21	1.48	1.84	0.66	-0.017	1.09
Full-Time Pilots	7483	1086.5	-0.95	28.9	3.49	1.88	-0.012	0.91	2.44	0.65	-0.265	0.81
Part-Time Pilots	7365	1928	-1.16	24.	4.14	2.58	0.058	2.02	1.67	0.93	-0.46	1.31
Nonpilots	7186	1150	-6.16	26.1	3.83	2.38	-0.44	1.21	1.81	1.06	0.369	0.91

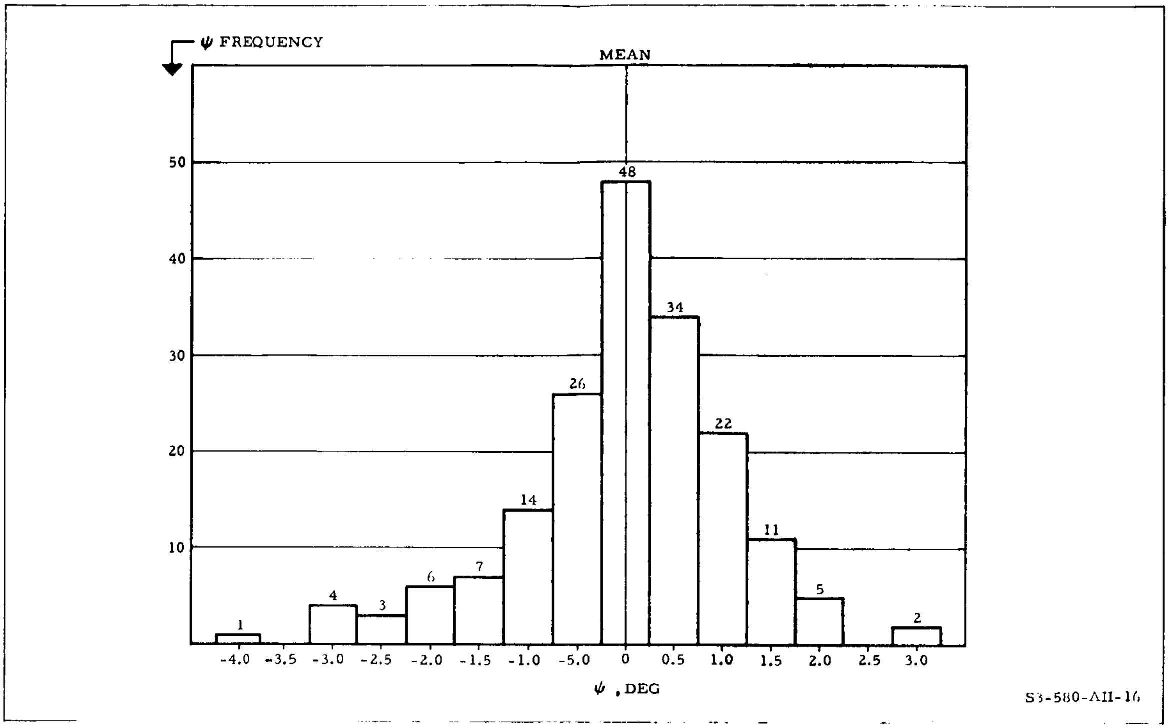

Figure AII-16. Histogram of ψ at Touchdown

An effort was made to evaluate the effectiveness of the runway centerline. Each pilot flew two sets of runs, the first set without a centerline, and the second set with a centerline. Values of Y_I at touchdown were observed (Table AII-3) and a slight trend was evident in favor of the centerline. All pilots also indicated a preference for the centerline because it provides a further reference for lateral maneuvers and adds confidence that the Y_I position is proper for touchdown.

Conclusions made as a result of the analog simulation are as follows:

1. The display as simulated provides an excellent source of information for landing and short-range navigational tasks. A heads-up display would most certainly be a profitable refinement.

2. The range marker should be a measure of direct ground range to the takeoff end of the runway.

3. The altitude marker should be referenced to the horizon line rather than to scope center. This provides flare information.

4. A runway centerline is a useful addition to the display.

5. The airplane cursor should be etched above the scope center to provide more area beneath it for runway lines, horizon, etc.

6. The angle marker is more functional if it represents $h + \psi - \psi_o$ rather than h .

7. The sink rate marker was unnecessary. Sink rate is roughly intuitive from h marker movement.

8. The simulation did not allow for flying past the runway. Had this been included it would have been possible to perform interesting navigational maneuvers based on pilot use of range and heading markers.

9. Glide Slopes could easily be set up and maintained using either range, altitude, and sink rate information or velocity circle and runway threshold line.

10. Pilots and nonpilots (familiar with pilot task) adjust readily to the display. They all make fairly consistent safe landings after 1/2 - 1 hr at the controls.

11. The display appears to lend itself well to the applications of STOL aircraft and near-sonic-transport landing missions.

Table AII-3. Effect of Runway Centerline Inclusion on Y_I

	Without Centerline		With Centerline	
	Mean	Standard Deviation	Mean	Standard Deviation
Y_I Ft	-7.4	22.9	1.0	27.0

C. EQUATIONS USED IN THE ANALOG SIMULATION

A list of equations used for the Analog Simulation follows. Coordinates are shown in Figures AII-17 through AII-19.

1. Aircraft Equations

a. Longitudinal

$$\dot{\gamma} = \alpha \left[0.000783 V_o C_D + \frac{0.563}{V_o} \gamma \right] + 0.0449 V_o C_L - \frac{1845}{V_o}$$

$$\ddot{\theta} = 0.00217 V_o^2 C_M - 0.00497 V_o \dot{\theta}$$

$$\alpha = \theta - \gamma$$

$$C_D = 0.0713 + 0.052 C_L^2$$

$$C_{L_o} = 0.321 - 0.200\ \alpha + 0.010 \delta_E$$

$$C_{M_o} = 0.113 - 0.0396\ \alpha - 0.0338 \delta_E$$

$$\delta_E = K_{\delta_E} \left[\delta_{S_E} + K_{TRIM} \int_o^t \delta_{TRIM} dt \right]$$

$$\dot{X} = V_o = 200 \text{ ft/sec}$$

$$K_{\delta_E} = 2 \text{ deg/in.}$$

$$K_{TRIM} = 0.25 \text{ deg/sec}$$

$$\delta_{TRIM} = \begin{cases} 1 & \text{with trim button on} \\ 0 & \text{with trim button off} \end{cases}$$

b. Lateral

$$\frac{\phi}{\delta_A} = \frac{1}{S^2 + 3.3S + 0.00033}$$

$$\dot{\psi} = \frac{32.2}{V_o} \phi$$

$$\delta A = \begin{cases} K_{\delta_A} \left[\delta_{S_A} - 1/8 \text{ in.} \right], & \left| \delta_{S_A} \right| > 1/8 \text{ in.} \\ 0, & \left| \delta_{S_A} \right| < 1/8 \text{ in.} \end{cases}$$

$$K_{\delta_A} = 2 \text{ deg/in.}$$

$$\dot{y} = \dot{Y}_h - \dot{h} \frac{\phi}{57.3}$$

c. Ground Effect

$$C_L = \begin{cases} (-0.004h + 1.2)\, C_{L_o}, & h < 50 \text{ ft} \\ C_{L_o}, & h > 50 \text{ ft} \end{cases}$$

$$C_M \begin{cases} (0.004\, h + 0.8)\, C_{M_o}, & h < 50 \text{ ft} \\ C_M = C_{M_o}, & h > 50 \text{ ft} \end{cases}$$

2. Geometric Equations

$$\dot{h} = \frac{V_o \gamma}{57.3} + \dot{Y}_h \frac{\phi}{57.3}$$

$$\dot{Y}_h = -V_{W/E} \cos \psi$$

$$\left. \begin{aligned} \sin \psi &= \psi - \frac{\psi^3}{6} \\ \cos \psi &= 1 - \frac{\psi^2}{2} \end{aligned} \right\} \quad \psi \text{ in radians}$$

$$\dot{X}_I = -V_o \cos \psi$$

$$\dot{Y}_I = V_o \sin \psi - V_{W/E}$$

$$X_I = \int \dot{X}_I \, dt$$

$$Y_I = \int \dot{Y}_I \, dt$$

$$X = X_I \cos \psi - Y_I \sin \psi$$

$$Y = -Y_I \cos \psi - X_I \sin \psi$$

3. Display Equations

a. Runway Lines

$$A = -K_\theta \left[\psi + \frac{Y_I}{X_I} - \phi \left(\theta - \frac{h}{X_I} \right) \right]$$

$$C = -K_\theta \left[\psi + \frac{Y_I}{X_I - a} - \phi \left(\theta - \frac{h}{X_I - a} \right) \right]$$

$$E = -K_\theta \left[\psi + \frac{Y_I}{X_I - \ell} - \phi \left(\theta - \frac{h}{X_I - \ell} \right) \right]$$

$$J = -K_\theta \left[\theta + \frac{h}{X_I} + \phi \left(\psi + \frac{Y_I}{X_I} \right) \right]$$

$$K = -K_\theta \left[\theta + \frac{h}{X_I - a} + \phi \left(\psi + \frac{Y_I}{X_I - a} \right) \right]$$

$$L = -K_\theta \left[\theta + \frac{h}{X_I - 1} + \phi \left(\psi + \frac{Y_I}{X_I - \ell} \right) \right]$$

$$B = K_\theta \frac{b}{X_I}$$

$$D = K_\theta \frac{b}{X_I - a}$$

$$F = K_\theta \frac{b}{X_I - \ell}$$

$$G = \phi B$$

$$H = \phi D$$

$$I = \phi F$$

b. Alternate Runway Lines

$$A = K_\theta \left(\frac{Y}{X} + \frac{h}{X} \phi \right)$$

$$B = K_\theta \left(\frac{b}{R_h} \right)$$

$$C = K_\theta \left(\frac{Y + a\,\psi + h\phi}{X - a} \right)$$

$$D = K_\theta \left(\frac{b}{R'_h} \right)$$

$$E = K_\theta \left(\frac{Y + 1\,\psi + h\phi}{X - \ell} \right)$$

$$F = K_\theta \left(\frac{b}{R''_h} \right)$$

$$G = B(\phi + \psi)$$

$$H = D(\phi + \psi)$$

$$I = F(\phi + \psi)$$

$$L = E(\phi + \psi) - K_\theta \left[\frac{Y\,\psi + h}{X - \ell} + \theta \right]$$

$$K = C(\phi + \psi) - K_\theta \left[\frac{Y\,\psi + h}{X - a} + \theta \right]$$

$$J = A(\phi + \psi) - K_\theta \left[\frac{Y\,\psi + h}{X} + \theta \right]$$

$$R_h = \sqrt{X^2 + Y^2}$$

$$R'_h = R_h - a$$

$$R''_h = R_h - 1$$

c. Velocity Circle

$$x_d = K_\theta \frac{\dot{y}}{V_o}$$

$$y_d = K_\theta \, \alpha$$

d. Alternate Velocity Circle

$$x_d = K_\theta \left[-\frac{\dot{Y}_I}{\dot{X}_I} - \psi + \phi \left(\theta + \frac{\dot{h}}{\dot{X}_I} \right) \right]$$

$$y_d = -K_\theta \left[\theta + \frac{\dot{h}}{\dot{X}_I} - \phi \left(-\frac{\dot{Y}_I}{\dot{X}_I} - \psi \right) \right]$$

e. Horizon Line

$$y_d = \phi \, x_d - K_\theta \theta$$

$$K_\theta = 15 \text{ in./rad}$$

f. Runway Bearing Marker

$$\zeta = \tan^{-1} \frac{Y}{X}$$

$$x_d = 2.25 \sin(\zeta + \psi)$$

$$y_d = 2.25 \cos(\zeta + \psi)$$

$$\Delta x_d = 0.5 \sin(\zeta + \psi)$$

$$\Delta y_d = 0.5 \cos(\zeta + \psi)$$

g. Range Dot

$$R_h = \sqrt{X^2 + Y^2}$$

$$x_d = 2.25 \sin(180 \text{ deg} - 5 \times 10^{-3} R_h)$$

$$y_d = -2.25 \cos(180 \text{ deg} - 5 \times 10^{-3} R_h)$$

h. Altitude Dot Operative for $X_I \leq (\ell + 1000)]$

x_d Same as for velocity circle

$$y_d = \begin{cases} \frac{h}{50} - K_\theta \theta, & h \leq 100' \\ 2, & h \Sigma 100' \end{cases}$$

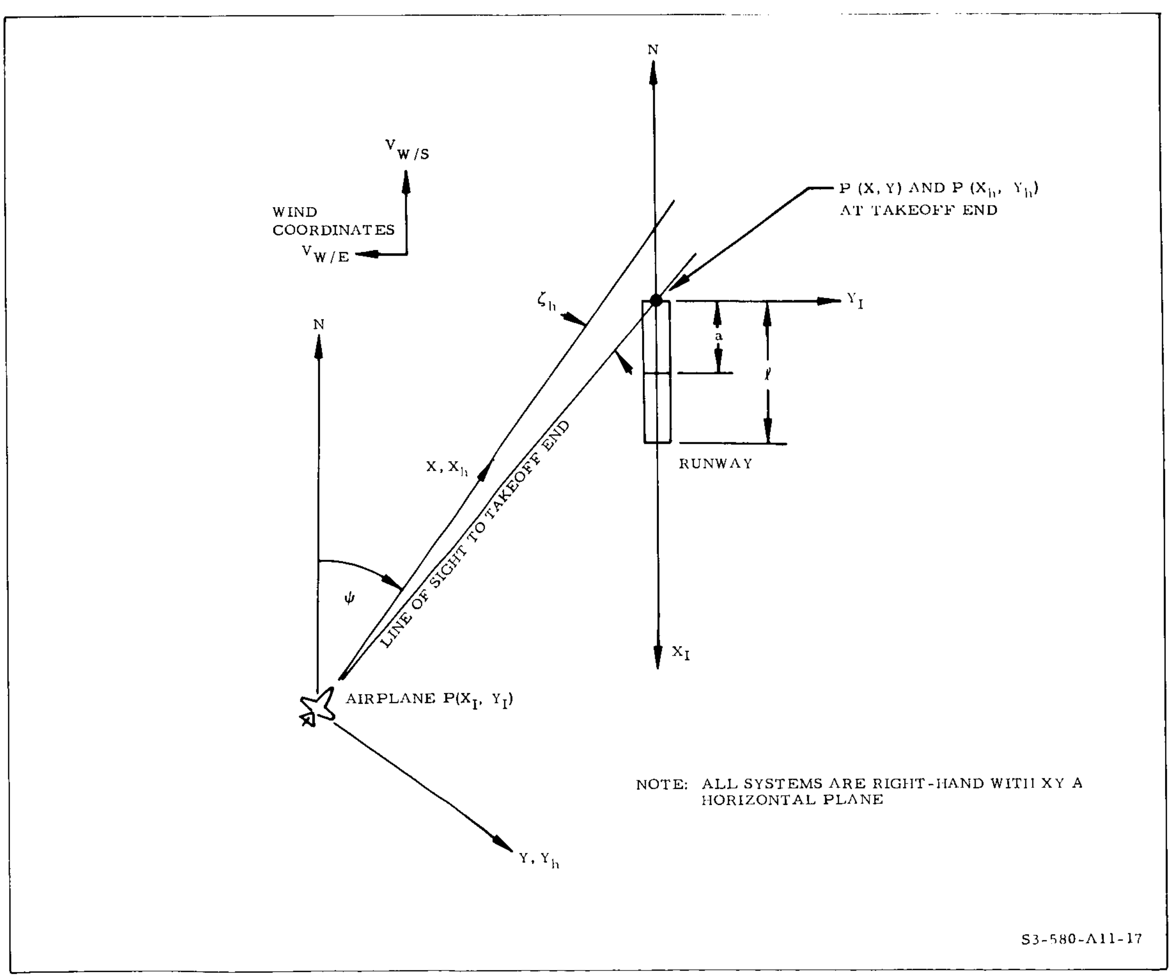

Figure AII-17. Horizontal Plane Coordinate System

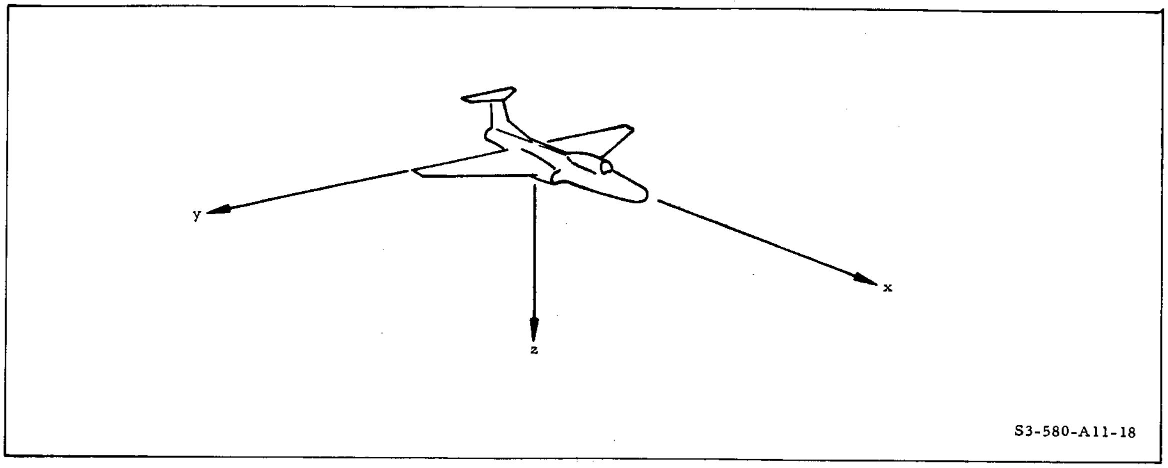

Figure AII-18. **Aircraft Coordinates**

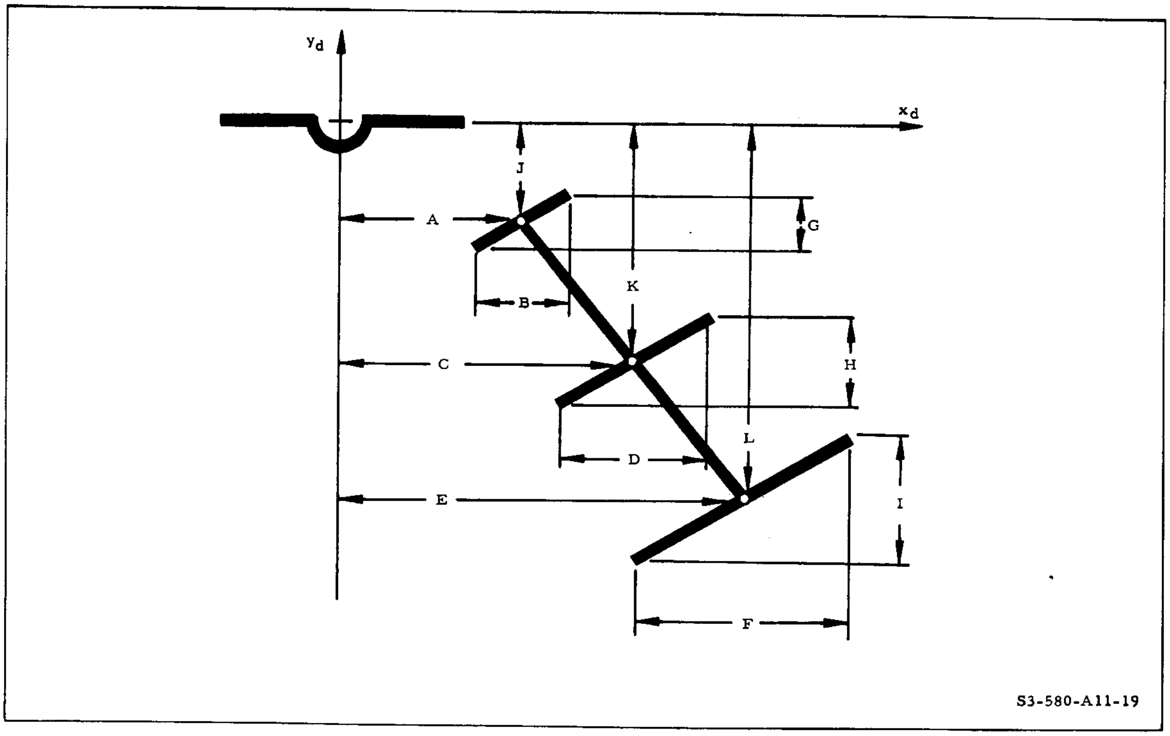

Figure AII-19. Runway Line Parameters

APPENDIX III. ERROR ANALYSIS

The errors affecting system performance can be grouped into three categories. These are: (1) sensor errors, (2) computation errors, and (3) errors arising from the use of approximations in the equations which generate the display. Illustrations (Figures AIII-1 through AIII-8) pertaining to error analysis are presented at the end of this section.

A. SENSOR ERRORS

Sensor errors are viewed as to how they affect the knowledge of position in an inertial coordinate system. For the purposes of this task, the error analysis done on inertial navigators in Section V is sufficient.

1. Radar

The expression relating slant range to inertial distances from the target being tracked by a radar (see Figure AIII-1) is given by

$$\begin{bmatrix} X_I \\ Y_I \\ Z_I \end{bmatrix} = \begin{bmatrix} \psi \end{bmatrix} \begin{bmatrix} \theta \end{bmatrix} \begin{bmatrix} \phi \end{bmatrix} \begin{bmatrix} \zeta \end{bmatrix} \begin{bmatrix} \eta \end{bmatrix} \begin{bmatrix} R \\ 0 \\ 0 \end{bmatrix}$$

if the attitude and heading sensor has a gimbal order of Z - Y - X and the radar antenna has a gimbal order of Z - Y from outside in. The matrices are defined as

$$\begin{bmatrix} \psi \end{bmatrix} = \begin{bmatrix} \cos\psi & -\sin\psi & 0 \\ \sin\psi & \cos\psi & 0 \\ 0 & 0 & 1 \end{bmatrix}, \quad \begin{bmatrix} \theta \end{bmatrix} = \begin{bmatrix} \cos\theta & 0 & \sin\theta \\ 0 & 1 & 0 \\ -\sin\theta & 0 & \cos\theta \end{bmatrix}$$

$$\begin{bmatrix} \phi \end{bmatrix} = \begin{bmatrix} 1 & 0 & 0 \\ 0 & \cos\phi & -\sin\phi \\ 0 & \sin\phi & \cos\phi \end{bmatrix}, \quad \begin{bmatrix} \zeta \end{bmatrix} = \begin{bmatrix} \cos\zeta & -\sin\zeta & 0 \\ \sin\zeta & \cos\zeta & 0 \\ 0 & 0 & 1 \end{bmatrix}$$

and $$\begin{bmatrix} \eta \end{bmatrix} = \begin{bmatrix} \cos\eta & 0 & \sin\eta \\ 0 & 1 & 0 \\ -\sin\eta & 0 & \cos\eta \end{bmatrix}$$

Thus the inertial coordinates, as defined in Figure AIII-1 can be expressed as:

$$X_I = R \left\{ \cos\psi \left[\cos\theta \cos\zeta \cos\eta + \sin\theta (\sin\phi \sin\zeta \cos\eta - \cos\phi \sin\eta) \right] - \sin\psi \left[\cos\phi \sin\zeta \cos\eta + \sin\phi \sin\eta \right] \right\}$$

$$Y_I = R \left\{ \sin\psi \left[\cos\theta \cos\zeta \cos\eta + \sin\theta (\sin\phi \sin\zeta \cos\eta - \cos\phi \sin\eta) \right] + \cos\psi \left[\cos\phi \sin\zeta \cos\eta + \sin\phi \sin\eta \right] \right\}$$

$$Z_I = R \left\{ -\sin\theta \cos\zeta \cos\eta + \cos\theta (\sin\phi \sin\zeta \cos\eta - \cos\phi \sin\eta) \right\}$$

Expressions for the errors in these quantities as a function of the measured variable are:

$$\begin{aligned}
\Delta X_I = {} & - Y_I \Delta\psi + \frac{X_I}{R} \Delta R + R \Big\{ \Delta\theta \Big[-\sin\theta \cos\zeta \cos\eta \\
& + \cos\theta (\sin\phi \sin\zeta \cos\eta - \cos\phi \sin\eta) \Big] \\
& + \Delta\zeta \Big[-\cos\theta \sin\zeta \cos\eta + \sin\theta \sin\phi \cos\zeta \cos\eta \\
& + \cos\phi \cos\eta \cos\zeta \Big] + \Delta\eta \Big[-\cos\theta \cos\zeta \sin\eta \\
& + \sin\theta (-\sin\phi \sin\zeta \sin\eta - \cos\phi \cos\eta) - \cos\phi \sin\zeta \sin\eta \\
& + \sin\phi \cos\eta \Big] + \Delta\phi \Big[\sin\theta (\cos\phi \sin\zeta \cos\eta + \sin\phi \sin\eta) \\
& + \cos\phi \sin\eta - \sin\phi \sin\zeta \cos\eta \Big] \Big\}
\end{aligned}$$

$$\begin{aligned}
\Delta Y_I = {} & X_I \Delta\psi + \frac{Y_I}{R} \Delta R + R \Big\{ \Delta\theta \Big[-\sin\theta \cos\zeta \cos\eta \\
& + \cos\theta (\sin\phi \sin\zeta \cos\eta \ \cos\phi \sin\eta) \Big] + \Delta\zeta \Big[-\cos\theta \sin\zeta \cos\eta \\
& + \sin\theta \sin\phi \cos\zeta \cos\eta + \cos\phi \cos\eta \cos\zeta \Big] \\
& + \Delta\eta \Big[-\cos\theta \cos\zeta \sin\eta + \sin\theta (-\sin\phi \sin\zeta \sin\eta \\
& - \cos\phi \cos\eta) - \cos\phi \sin\zeta \sin\eta + \sin\phi \cos\eta \Big] \\
& + \Delta\phi \Big[\sin\theta (\cos\phi \sin\zeta \cos\eta + \sin\phi \sin\eta) + \cos\phi \sin\eta \\
& - \sin\phi \sin\zeta \cos\eta \Big] \Big\}
\end{aligned}$$

comparison of the combination of the various geometries and computational schemes. As such, it is included in Section A, III, C.

An analysis of altitude errors was conducted for the generalized geometries involved. The geometry is shown in figure AIII-1. For the two transponder case, altitude must be determined independently. This is because three independent measurements are needed to locate a point in space. Two of these are acquired by using DME, the third by using an altimeter.

For the four transponder case, where the fourth transponder is not in the plane of the other three, the error is approximately

$$|\Delta Z_A| \approx \frac{|\Delta R|}{n}\sqrt{R_c^{\,2} + R_d^{\,2}}$$

$$\approx \frac{|\Delta R|}{n}\sqrt{2}\;|X_A + \ell|$$

and if Δ R is taken to be 5 then

$$|\Delta Z_A| \approx \frac{7\,|X_A + \ell|}{n} \qquad \text{(AIII-1)}$$

For the three transponder case

$$|\Delta Z_A| \approx \frac{|\Delta R|}{\sqrt{2}}\left|\frac{X_A}{Z_A + h_t}\right|\left|\frac{X}{\ell}\right| \qquad \frac{X_A}{\ell} >> 1 \qquad \text{(AIII-2)}$$

If the best value of ℓ is used so that the $\frac{X_A}{\ell} \approx 1$, the error is always larger than

$$|\Delta Z_A| \geq \frac{|\Delta R|}{2(Z_A + h_t)}\sqrt{R_a^{\,2} + R_b^{\,2}}$$

$$\geq \frac{|\Delta R|}{|Z_A + h_t|}\sqrt{2}\;\ell \qquad \text{(Minimum possible)}$$

The preceding equations assume the following:

$$\frac{X_A + \ell}{n} \quad \text{is large}$$

$$\sqrt{R_a^{\;2} + R_c^{\;2}} \approx \sqrt{2}\; X_A, \; \frac{XA}{\ell} >> 1$$

$$\sqrt{R_a^{\;2} + R_b^{\;2}} \approx 2\sqrt{2}\; \ell, \; \frac{XA}{\ell} \approx 1$$

Evaluating these expressions for a low approach, the following conditions are reasonable: X_A = -15000 ft, ℓ = 5000 ft, n = 200 ft, Z_A = 500 ft, and h_t = 0 ft. Equation AIII-1 yields $|\Delta Z_A|$ = 350 ft, and AIII-2 $|\Delta Z_A|$ = 350 ft. Thus it can be seen that on this approach, the error is at least 40 percent of the altitude.

For all reasonable transponder geometries, in or near the plane of the airport, the altitude error is proportional to range and inversely proportional to approach angle. Altitude cannot be found accurately enough with this system.

3. Angle Sensors

The effects of these types of errors are discussed in Appendix II.

B. COMPUTATION ERRORS

Computational errors were not investigated because this would have entailed, among other things, outlining the actual program that would be used with the digital computer. This latter task was beyond the scope of this contract. In any event, the computer specification (refer to Appendix VII) calls for a word length of 20 bits. This is equivalent to between 5 and 6 decimal places. It is felt that this should lead to great enough accuracy since there are no iterative type or series solutions to be made and the output need only have between 3 and 4 decimal place accuracy.

C. ERRORS ARISING FROM APPROXIMATIONS

1. Radar

No work was expended in this area because it was discovered, prior to this phase of the study, that radar was not accurate enough for this application.

2. DME

An error analysis was performed on some of the approximate equations which relate DME sensed parameters to inertial coordinates. (Refer to Appendix I) This was done for the two, and the recommended four ground transceiver geometries. In both cases, separate altimeters are assumed. In the case of four transceivers, two altimeters are assumed, although this does not greatly affect the results.

The following set of equations describes the errors that arise with two transceivers:

$$\Delta X_A = \frac{\partial X_A}{\partial R_a} \Delta R_a + \frac{\partial X_A}{\partial R_b} \Delta R_b + \frac{\partial X_A}{\partial h} \Delta h \qquad \text{(AIII-3)}$$

$$\Delta Y_A = \frac{\partial Y_A}{\partial R_a} \Delta R_a + \frac{\partial Y_a}{\partial R_b} \Delta R_b \qquad \text{(AIII-4)}$$

$$\Delta Z_A = \frac{\partial Z_a}{\partial h} \Delta h \qquad \text{(AIII-5)}$$

The set for four transceivers is

$$\Delta X_A = \frac{\partial X_A}{\partial R_a} \Delta R_a + \frac{\partial X_A}{\partial R_b} \Delta R_b + \frac{\partial X_A}{\partial R_c} \Delta R_c + \frac{\partial X_A}{\partial R_d} \Delta R_d \qquad \text{(AIII-6)}$$

$$\Delta Y_A = \frac{\partial Y_A}{\partial R_a} \Delta R_a + \frac{\partial Y_A}{\partial R_b} \Delta R_b + \frac{\partial Y_A}{\partial R_c} \Delta R_c + \frac{\partial Y_A}{\partial R_d} \Delta R_d \qquad \text{(AIII-7)}$$

$$\Delta Z_A = \frac{\partial Z_A}{\partial h_1} \Delta h_1 + \frac{\partial Z_A}{\partial h_2} \Delta h_2 \qquad \text{(AIII-8)}$$

The partial derivatives to be used in these two equations are given in Tables AIII-1 and AIII-2. These tables are read as follows: the variable in the numerator of the partial is at the left of the table, the parameters in the denominators of the partials are along the top of the table. Thus $\frac{\partial X_A}{\partial R_a}$ is the element of the table which is at the intersection of the X_A row and the R_a column.

Table AIII-1. Partial Derivatives for Two Transceivers and Altimeter (Equations AIII-3, AIII-4, and AIII-5)

	R_a	R_b	h
X_A	$\pm \frac{R_a}{2X_A}\left(1 + \frac{Y_A}{M}\right)$ *	$\pm \frac{R_b}{2X_A}\left(1 - \frac{Y_A}{M}\right)$ *	$\pm \left(\frac{Z_A + h_t}{X_A}\right)$ *
Y_A	$\frac{-Ra}{2m}$	$\frac{Rb}{2m}$	0
Z_A	0	0	-1

*These terms have the same sign as the quantity $(\dot{R}_a + \dot{R}_b)$.

Table AIII-2. Partial Derivatives for Four Transceivers and Two Altimeters (Equations AIII-6, AIII-7, and AIII-8)

	R_a	R_b	R_c	R_d	h_1	h_2
X_A	$\frac{-R_a}{4\ell}$	$\frac{-R_b}{4\ell}$	$\frac{R_c}{4\ell}$	$\frac{R_d}{4\ell}$	0	0
Y_A	$\frac{-R_a}{4m}$	$\frac{R_b}{4m}$	$\frac{R_c}{4m}$	$\frac{-R_d}{4m}$	0	0
Z_A	0	0	0	0	-1/2	-1/2

The results of these equations are shown in Figure AIII-6.

3. Display

The several sets of equations describing the display are listed in Appendix I. An error analysis of the preferred set follows.

For the runaway lines,

$$\Delta\left(\frac{y}{R}\right)_{D_i} = -\left[\psi - \sin\psi\cos\phi\left(\frac{X_i - X_A}{R_i}\right)\right] +$$

$$\left[\theta\ \phi - \sin\theta\sin\phi\cos\psi\left(\frac{X_i - X_A}{R_i}\right)\right] + \left[\left(\frac{Y_i - Y_A}{X_i - X_A}\right)\right.$$

$$\left. - \left(\frac{Y_i - Y_A}{R_i}\right)(\cos\psi\cos\phi + \sin\psi\sin\theta\sin\phi)\right]$$

$$+\left[\left(\frac{Z_i - Z_A}{X_i - X_A}\right)\phi - \left(\frac{Z_i - Z_A}{R_i}\right)\sin\phi\cos\theta\right]$$

$$\Delta\left(\frac{z}{R}\right)_{D_i} = \theta - \sin\theta\cos\phi\left[\cos\psi\left(\frac{X_i - X_A}{R_i}\right) + \sin\psi\left(\frac{Y_i - Y_A}{R_i}\right)\right]$$

$$+\left[\psi\ \phi - \sin\psi\sin\phi\left(\frac{X_i - X_A}{R_i}\right)\right] - \left[\left(\frac{Y_i - Y_A}{X_i - X_A}\right)\phi\right.$$

$$\left. - \left(\frac{Y_i - Y_A}{R_i}\right)\sin\phi\cos\psi\right] + \left[\left(\frac{Z_i - Z_A}{X_i - X_A}\right)\right.$$

$$\left. - \left(\frac{Z_i - Z_A}{R_i}\right)\cos\theta\cos\phi\right].$$

For the velocity circle, the errors are

$$\Delta\left(\frac{y}{R}\right)_{D_v} = -\left[\psi - \sin\psi\cos\phi\left(\frac{dX_A}{dR_\ell}\right)\right] + \left[\theta\ \phi - \sin\theta\cos\phi\cos\psi\left(\frac{dX_A}{dR_\ell}\right)\right]$$

$$+\left[\frac{dY_A}{dX_A} - \frac{dY_A}{dR_\ell}\left(\cos\psi\cos\phi + \sin\psi\sin\theta\sin\phi\right)\right]$$

$$+\left[\left(\frac{dZ_A}{dX_A}\right)\phi - \left(\frac{dZ_A}{dR_\ell}\right)\sin\phi\cos\theta\right]$$

$$\Delta\left(\frac{Z}{R}\right)_{D_V} = \theta - \sin\theta\cos\phi\left[\cos\psi\left(\frac{dX_A}{dR_\ell}\right) + \sin\psi\left(\frac{dY_A}{dR_\ell}\right)\right]$$

$$+\left[\psi\ \phi - \sin\psi\ \sin\phi\left(\frac{dX_A}{dR_\ell}\right)\right] - \left[\left(\frac{dY_A}{dX_A}\right)\ \phi\right.$$

$$\left. - \left(\frac{dY_A}{dR_\ell}\right)\sin\phi\cos\psi\right] + \left[\left(\frac{dZ_A}{dX_A}\right) - \left(\frac{dZ_A}{dR_\ell}\right)\cos\theta\cos\phi\right]$$

The dominant part of these errors comes from the terms $\frac{Y_i - Y_A}{X_i - X_A}$, $\frac{Z_i - Z_A}{X_i - X_A}$, $\frac{dY_A}{dX_A}$, and $\frac{dZ_A}{dX_A}$ when an approach is made at large angles to the runway heading. Figure AIII-7 shows this effect. Most of the error shows up as a velocity circle shift, but is reduced with the improved approximations up to approach angles of about 40 deg.

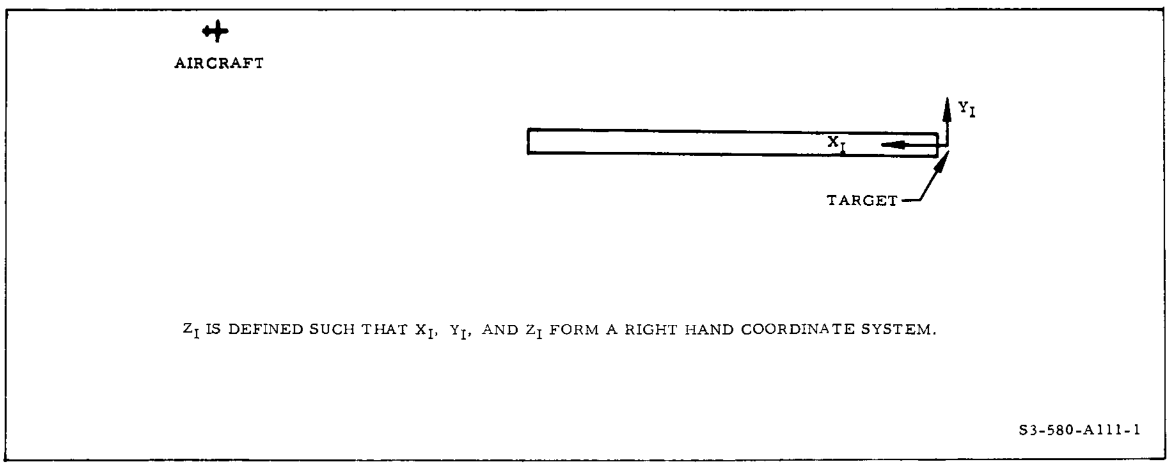

Figure AIII-1. Radar Target Geometry

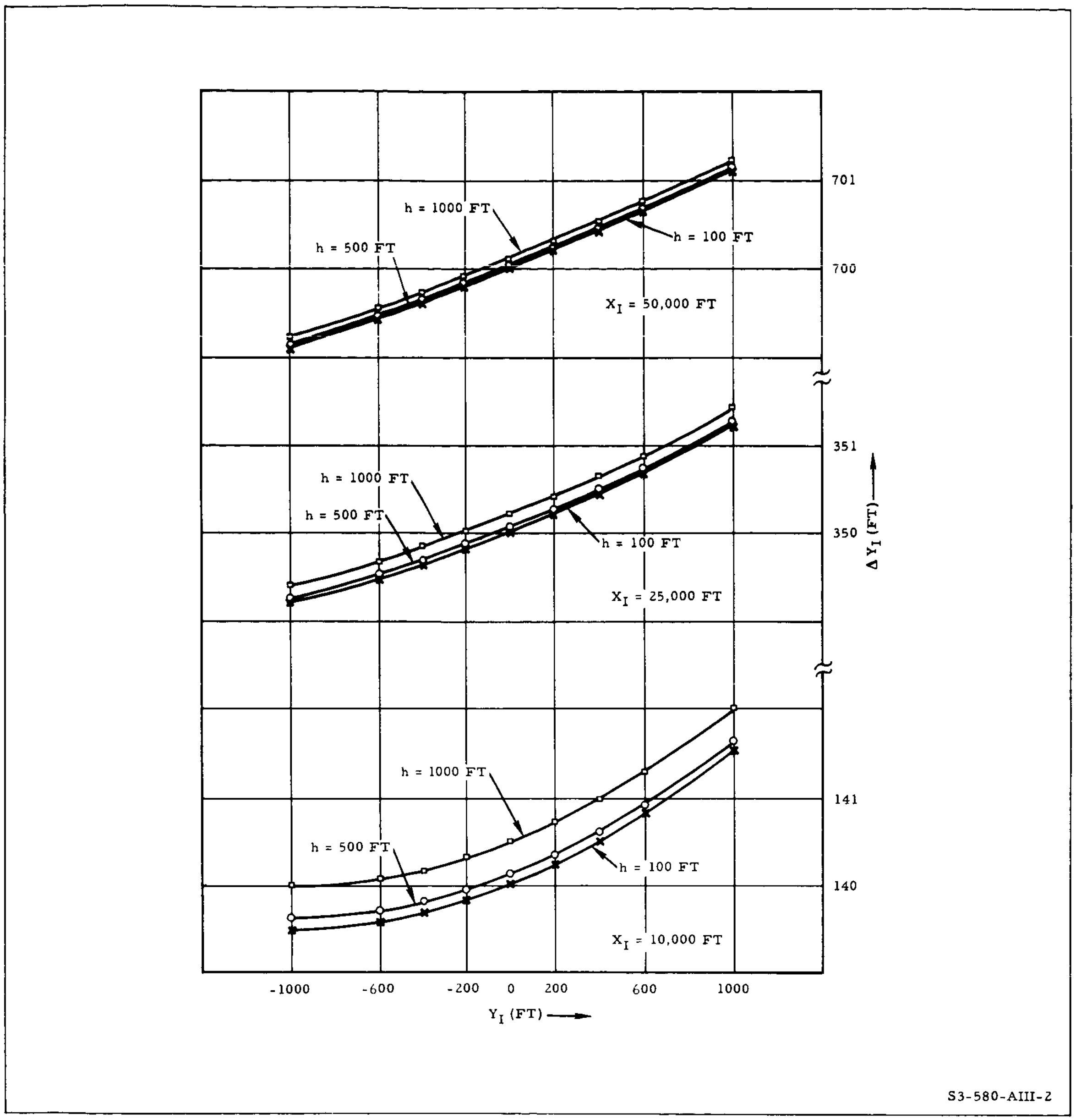

Figure AIII-2. Lateral Error - Radar

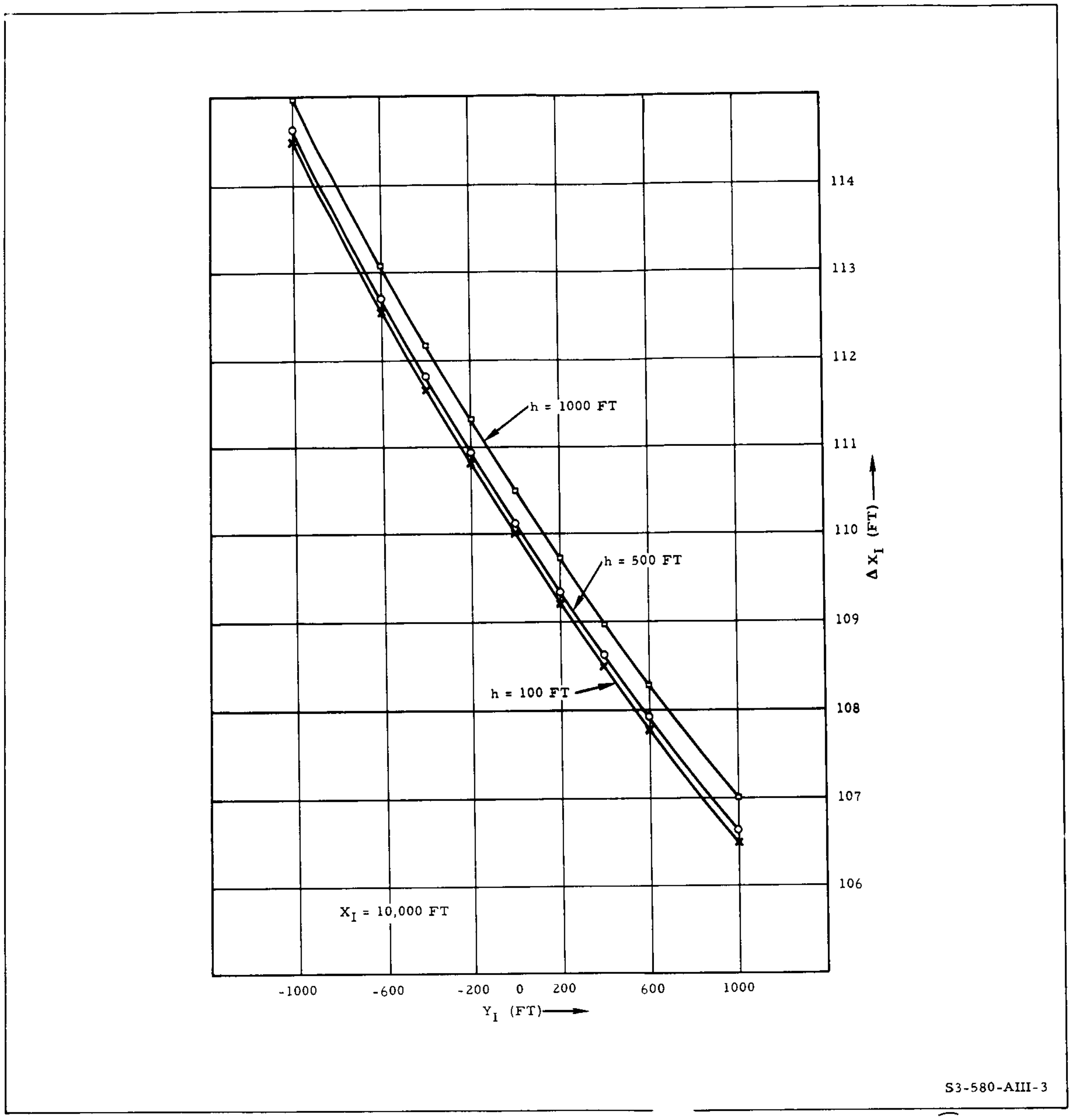

Figure AIII-3. Longitudinal Error - Radar X_i = 10,000 Ft

2. Diagnostic Test. A routine shall be provided which isolates malfunctions to subassemblies within the computer.

3. Subroutines. Subroutines shall be provided as specified by the procuring agency. Minimum requirements will be for sine, cosine, arctangent, double precision arithmetic, and a compiler.

b. Manuals

Those manuals necessary to program, install, maintain, and operate the computer and control panel shall be provided with the hardware.

c. Training

A training program shall be provided for at least one person in programing the computer and one person in maintaining the computer.

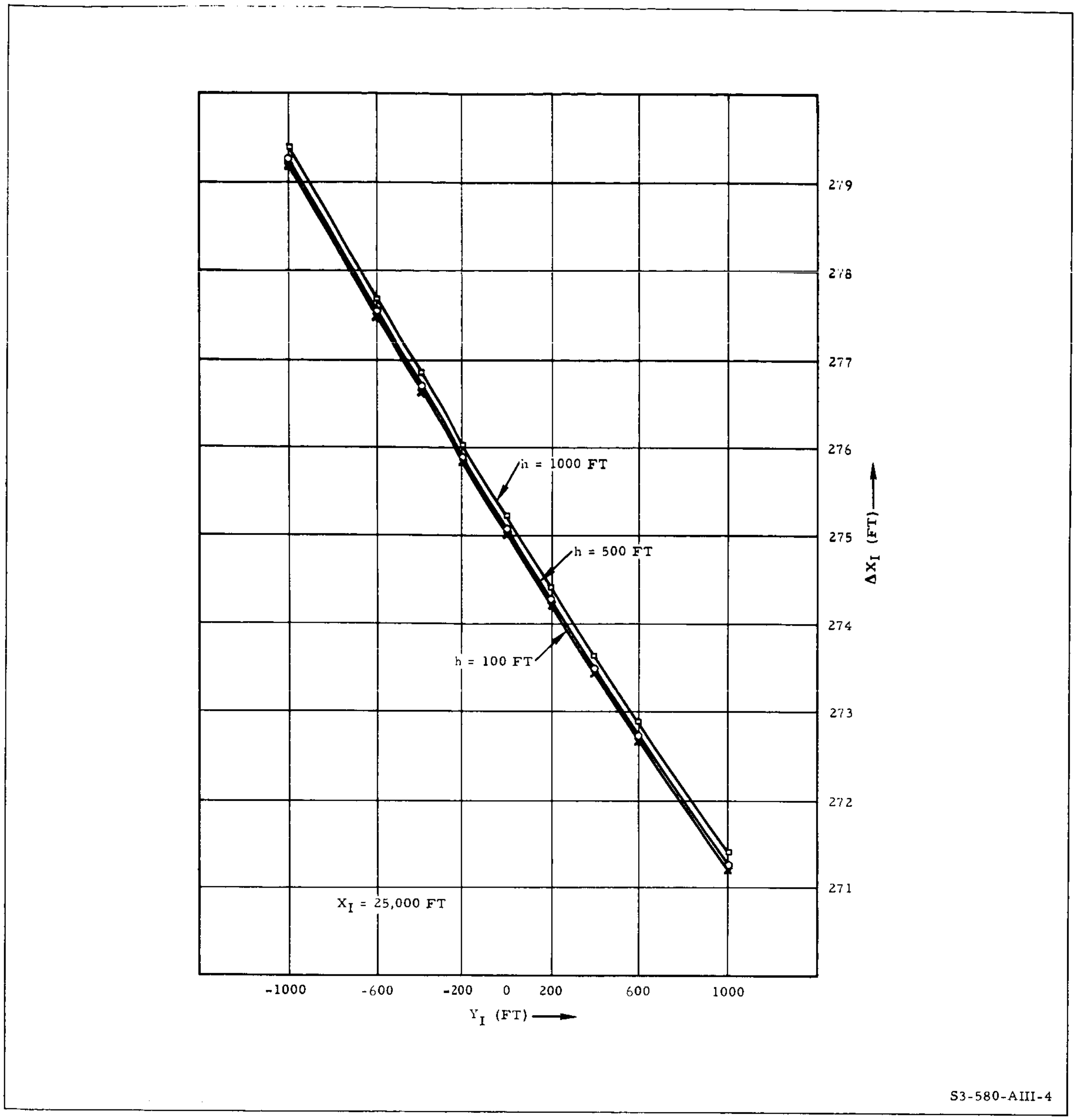

Figure AIII-4. Longitudinal Error - Radar X_i = 25, 000 Ft

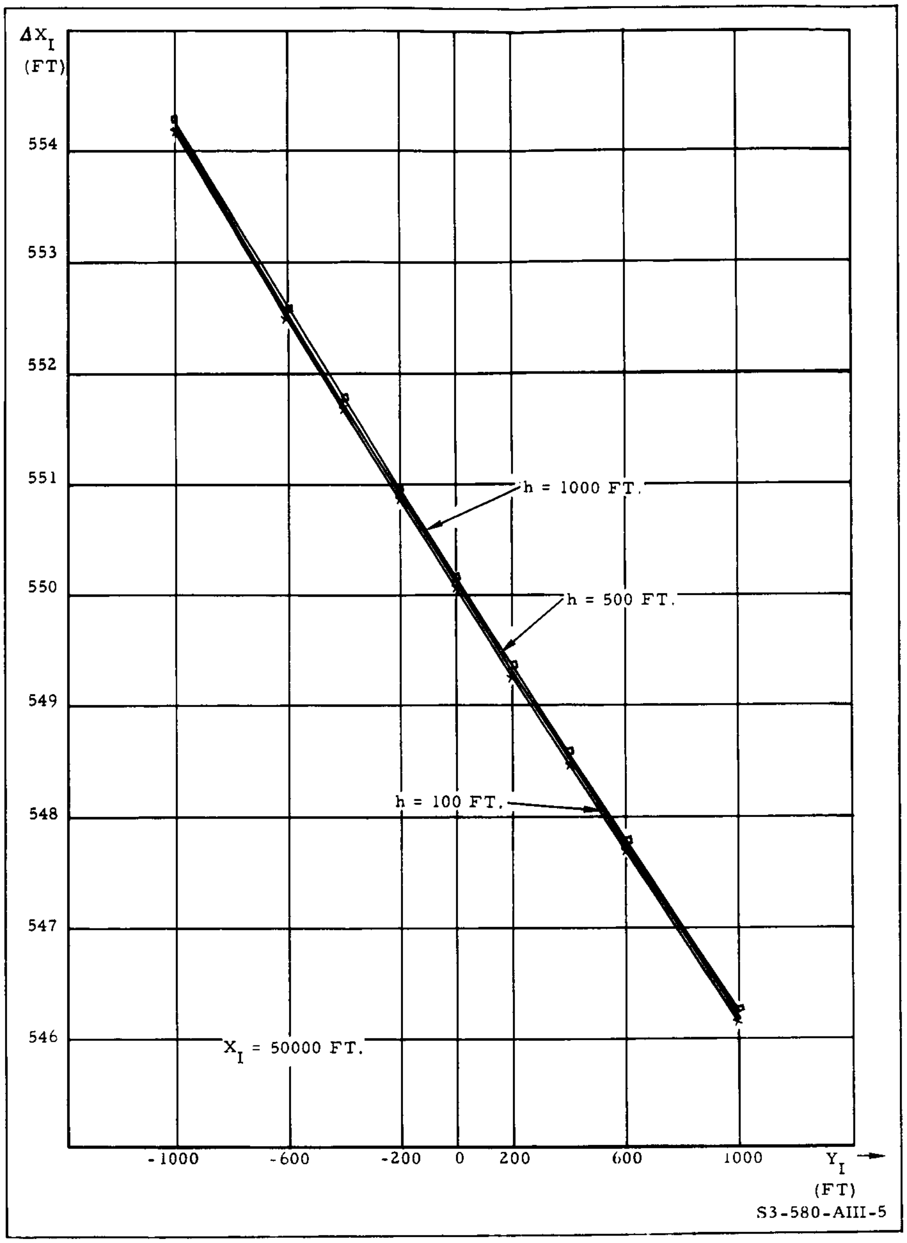

Figure AIII-5. Longitudinal Error - Radar X_i = 50,000 Ft

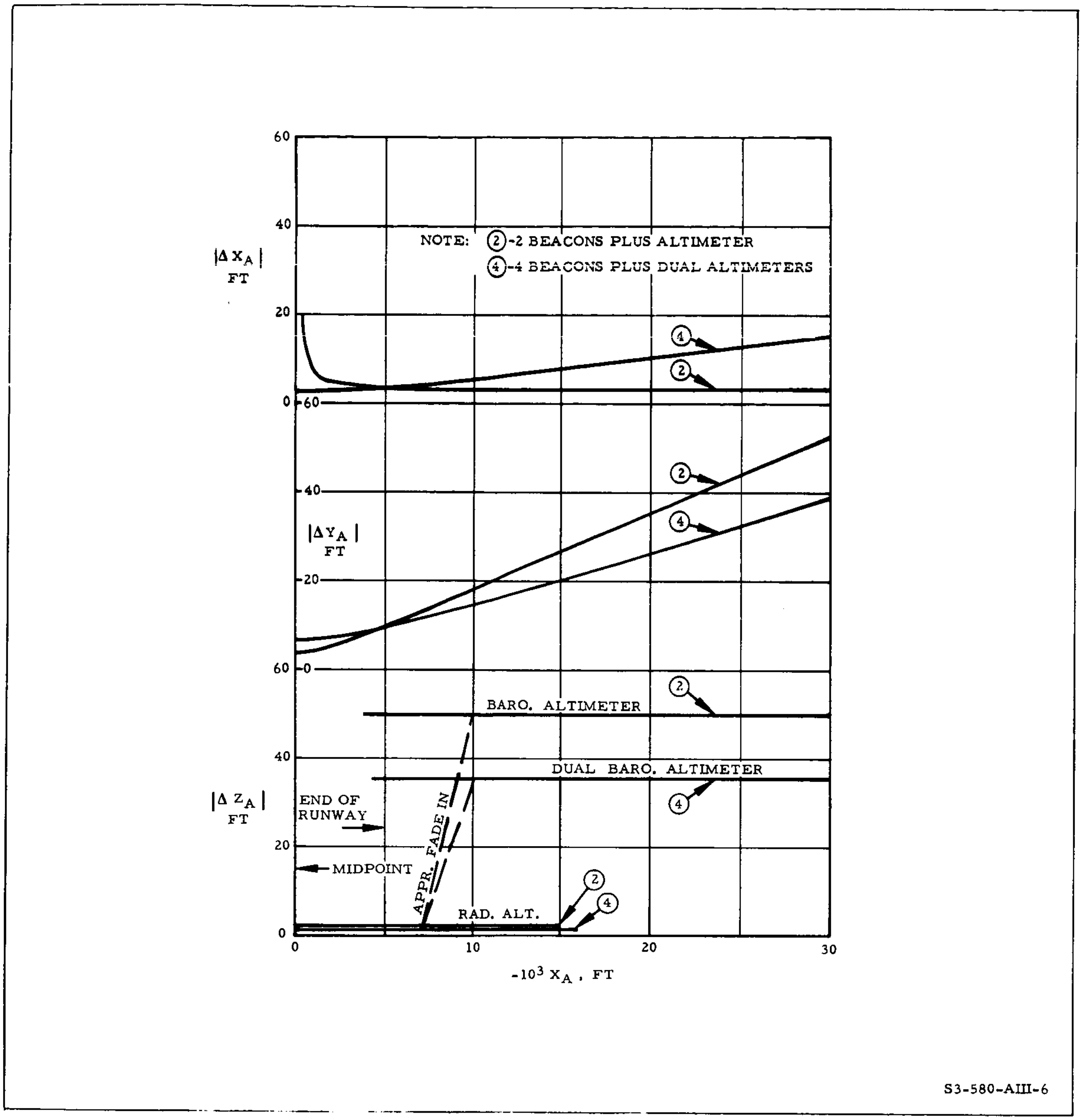

Figure AIII-6. One-Sigma Position Errors With Practical Beacon Geometry

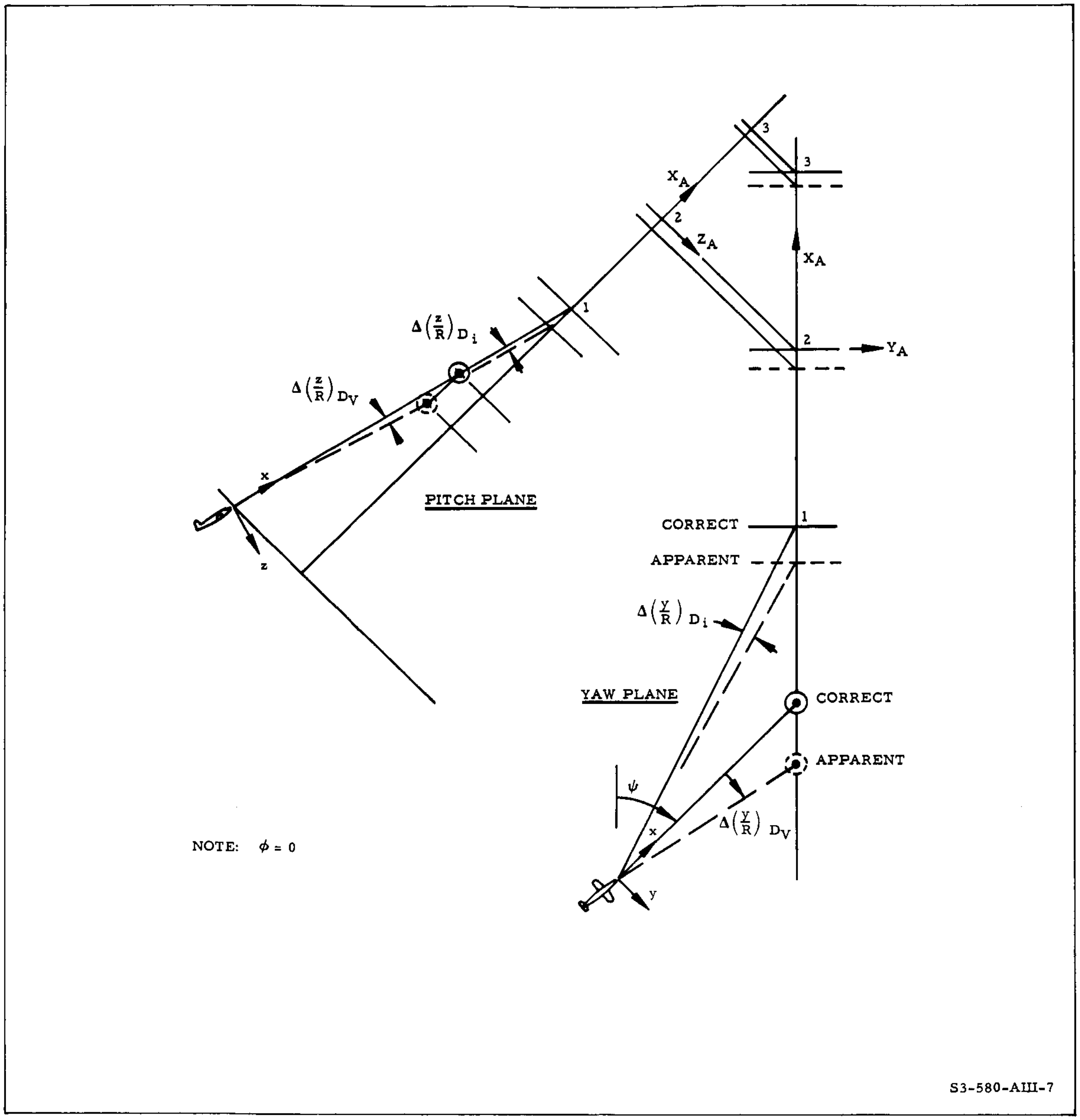

Figure AIII-7. Dominant Errors Caused by Approximate Display Formulas of Ref a.

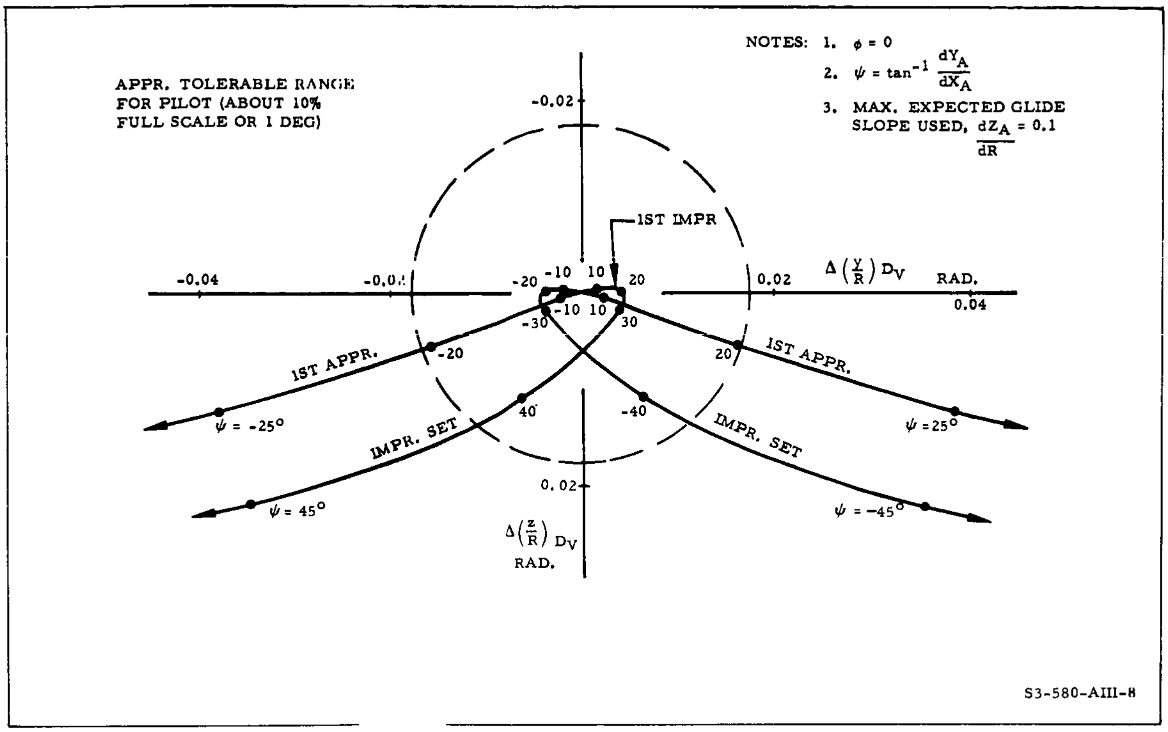

Figure AIII-8. Comparison of Dominant Errors for Display Direction Code

APPENDIX IV. MULTIPLEXER BREADBOARD

The results of investigations into circuit configurations suitable for generating the display from the outputs of the MARDAN computer were confirmed by breadboarding the circuitry for test. MARDAN outputs were simulated and six symbols were displayed on a CRT.

The circuitry for display generation can be grouped into three basic sections. The first is an algebraic and signal conversion section which converts d-c computer voltages to composite display generating signals. The second is a time sharing section which allows the six pieces of information to be displayed sequentially on the crt tube, while the third is an indicator unit which contains the crt tube and related horizontal, vertical, and intensity amplifiers. A simplified block diagram of the display showing the important subsections is illustrated in Figure A IV-1. Figure A IV-2 is the mechanization of the configuration indicated in the block diagram of Figure A IV-1. The multiplexer section is further detailed in Figure A IV-3.

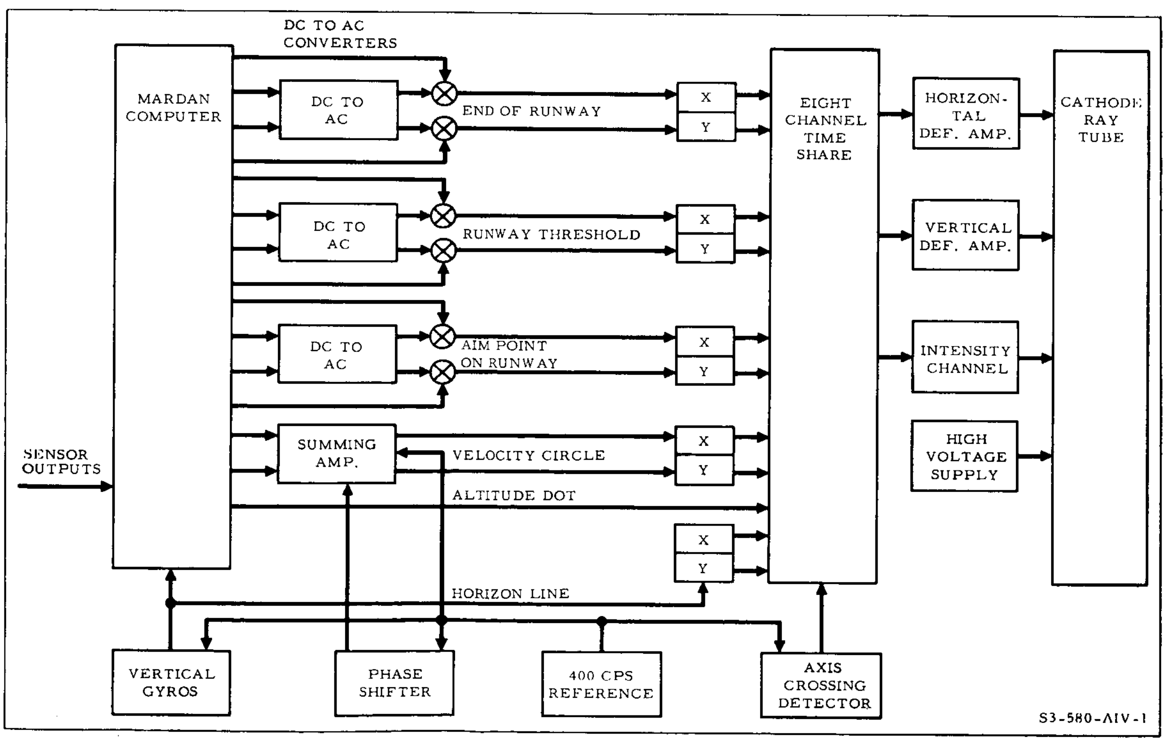

Figure AIV-1. Simplified Block Diagram of the Landing Display Configuration

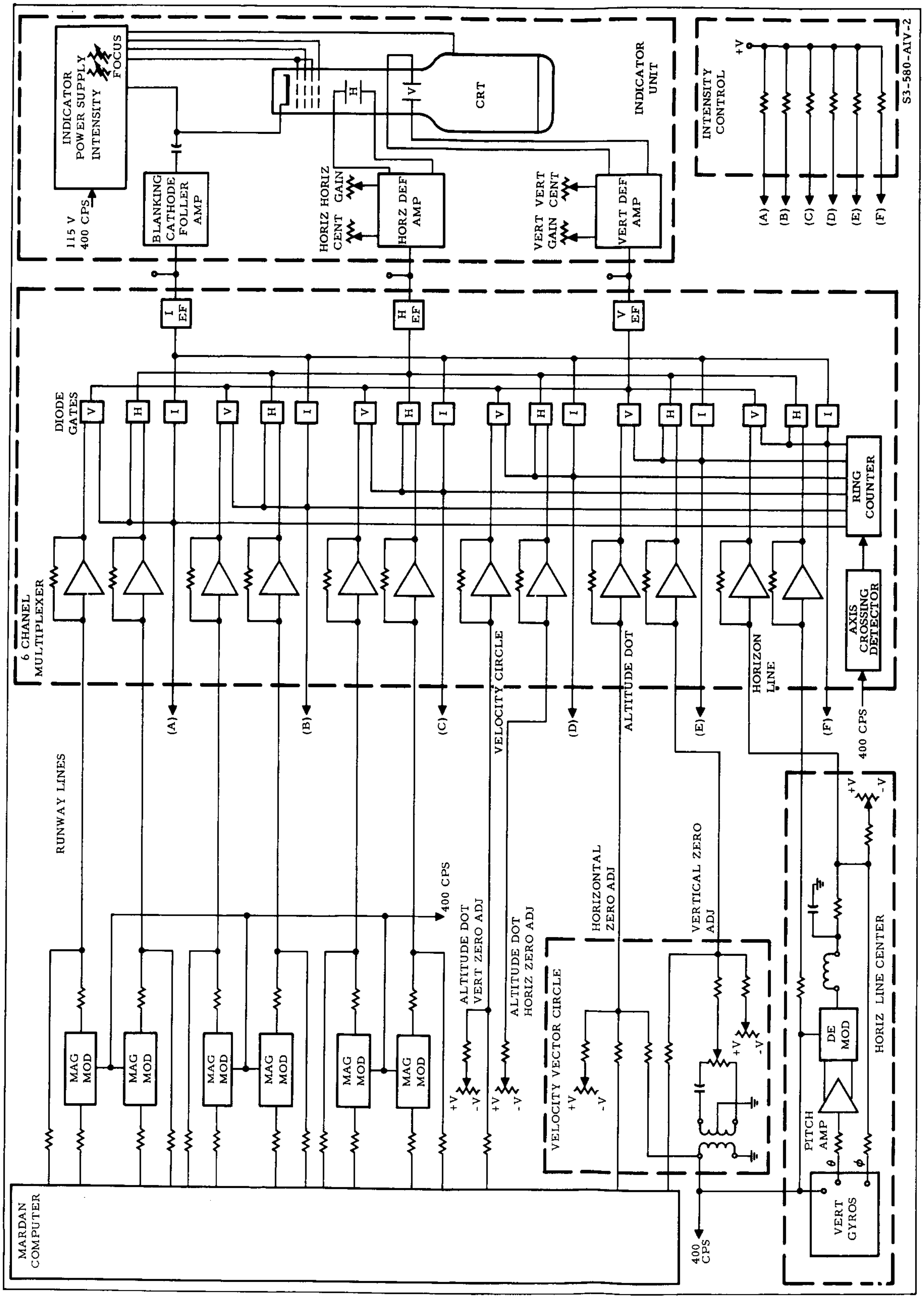

Figure AIV-2. Display Generating Circuitry

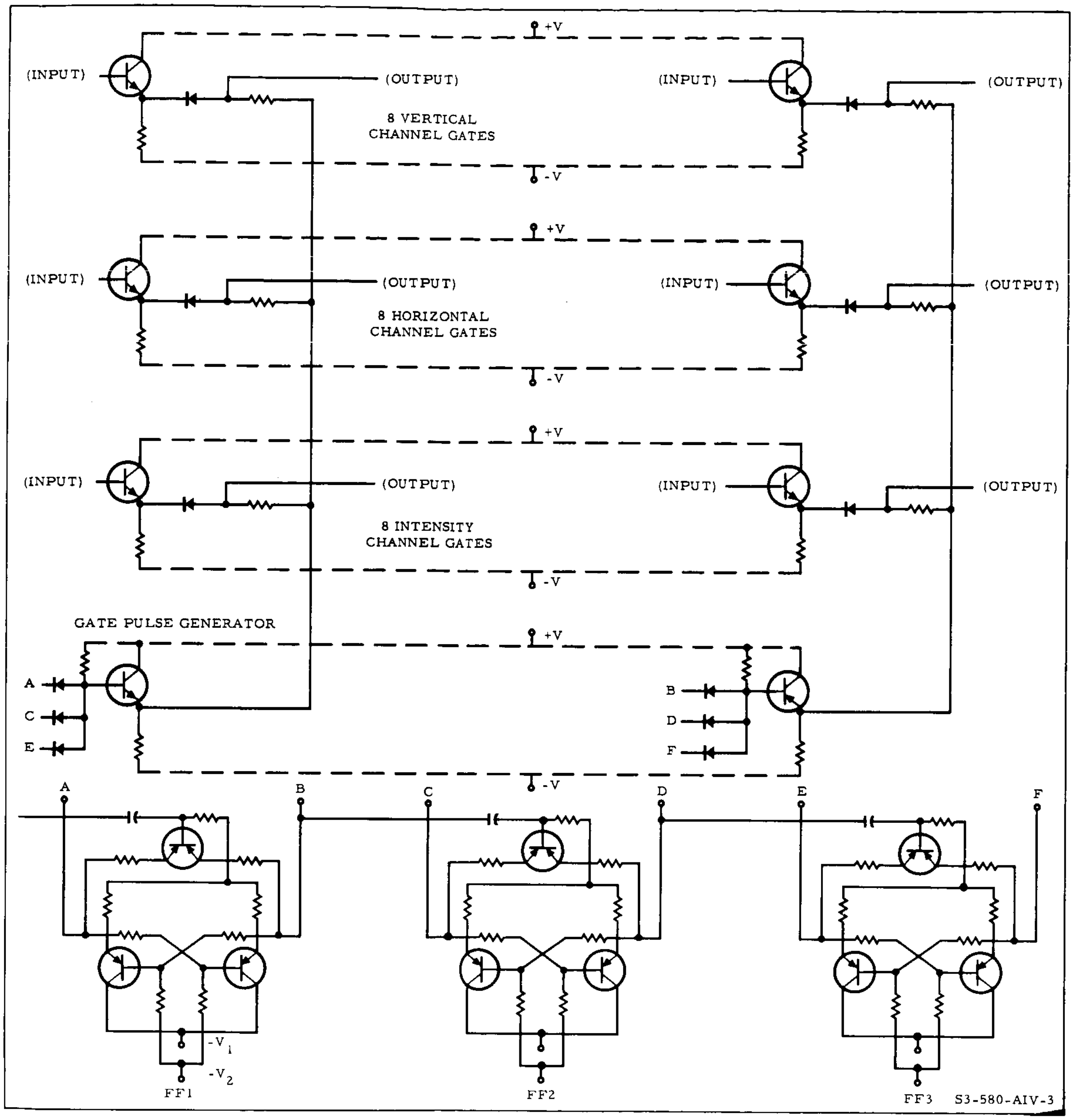

Figure AIV-3. Multiplexer Mechanization

The pitch gyro pickoff is fed into a phase inverter and isolation amplifier, if only an a-c computer exists, to obtain the double ended output required to drive a phase sensitive demodulator. Demodulation of the suppressed carrier signal gives a d-c voltage which defines the vertical location of the horizon line. A second-order L-C filter is used after the demodulator to ensure elimination of 800-cps ripple and produce a steady d-c signal. This would not be necessary if a d-c pickoff were used.

The d-c signal obtained from the pitch gyro is summed with the 400-cps suppressed carrier signal from the roll gyro pickoff. If a d-c pickoff were used the roll signal must be modulated. This composite signal is then time shared by the vertical deflection system. When the 400-cps reference signal is applied to the horizontal deflection system in conjunction with the composite vertical signal, the horizon line is generated.

For the runway lines, the computer voltages are converted to sinusoidally varying signals by the use of multiplying magnetic modulators. The magnetic modulator is a d-c to a-c converter which presents a linear relationship between a d-c input control signal and a-c output. The variable amplitude bipolar d-c voltage from the computer, when applied to the magnetic modulator, produces a corresponding variable amplitude phase reversing a-c output.

The use of magnetic modulators gives sinusoidal wave shapes with low harmonic distortion without frequency sensitive filtering circuits which would be required with chopper modulators.

To obtain the velocity circle, two constant amplitude carrier signals of 0 and 90 deg are summed with computer coordinate voltages. The MARDAN digital computer supplies the coordinate voltages which have a maximum amplitude of ± 10 v. These computer voltages are summed directly with the carrier signals and are commutated to the deflection system. The 400-cps carrier signal is sufficient for circle generation.

The altitude dot is defined by a d-c voltage to the vertical deflection system. This computed voltage is obtained directly from the MARDAN computer and is commutated to the deflection system. The horizontal deflection voltage is the same as that for the velocity circle.

Intensity control is provided for all pieces of information to maintain constant display intensity at the various writing rates. This is obtained by commutating a given pedestal voltage to the intensity channel simultaneously with each set of display signals.

The six sets of computed display information are commutated to the horizontal, vertical and intensity channels of the indicator by a three-pole eight-channel time-sharing multiplexer. The multiplexer is synchronized to the 400-cps line by an axis crossing detector. This results in a 2500-usec gate per channel with a repetition rate of 50 times/sec (based on an eight channel multiplexer).

The information display rate of 50 times/sec is sufficient to obtain a steady and accurate presentation on the CRT tube.

The indicator unit contains a horizontal, vertical and intensity channel amplifier, plus the related power supplies necessary for CRT tube operation. The vertical and horizontal amplifiers are d-c coupled and provide a gain necessary to give a deflection sensitivity of 1 in. per volt. A 5-in. crt tube with medium persistence phosphor was found to be best suited for this breadboard display mechanization.

Simple circuitry can be used for the display generation. No real interface problems were found to exist with this mechanization.

The 400-cps carrier voltage was found to be sufficient for display generation.

The indicator unit is the only component not available in the required form. However, this unit uses standard circuitry and techniques in CRT tubepresentations and can be easily designed.

APPENDIX V. LIST OF MANUFACTURERS CONTACTED

Included here is a complete list of the manufacturers and their addresses contacted in the course of this study. The list is divided into three parts. Part A includes those manufacturers from whom useful information was obtained. Much of this information is contained throughout this report. Part B includes those manufacturers who replied to the inquiries made but whose responses were not technically useful for this investigation. Part C lists those manufacturers who were contacted by mail but from whom no reply was received.

PART A

ADAGE, INC.
292 Main St.
Cambridge, Massachusetts

ADCOLE CORP.
186 Massachusetts Ave.
Cambridge, Massachusetts

ASTRODATA, INC.
240 E. Palais Rd.
Anaheim, California

ASTRONAUTICS CORPORATION OF AMERICA
3830 W. Wisconsin Ave.
Milwaukee 8, Wisconsin

ASTRO SPACE LABORATORIES INC.
2104 Memorial Pkwy S. W.
Huntsville, Alabama

COMPUTER CONTROL CO., INC.
Old Connecticut Path
Framingham, Massachusetts

CONTROL DATA CORP.
8100 34th Ave. S.
Minneapolis 20, Minnesota

CUBIC CORP.
5575 Kearny Villa Road
San Diego 23, California

DEL MAR ENGINEERING LABORATORIES
6901 W. Imperial Hwy.
Los Angeles 45, California

DYNAMIC SYSTEM ELECTRONICS
4420 E. Osborne
Phoenix, Arizona

DYNATRONICS, INC.
Box 2566, Highway 17-92
North Orlando, Florida

EL-TEK, INC.
13040 S. Cerise Ave.
Hawthorne, California

EMERSON ELECTRIC MANUFACTURING CO.
8100 Florissant Ave.
Saint Louis 36, Missouri

EPSCO, INC.
275 Massachusetts Ave.
Cambridge 39, Massachusetts

FERRANTI ELECTRIC, INC.
Plainview, Long Island, New York

GENERAL ELECTRIC CO.
Apparatus Sales
1 River Road
Schenectady 5, New York

GENERAL INSTRUMENT CORP.
Radio Receptor Division
Andrews Road
Hicksville, Long Island, New York

GENERAL PRECISION INC.
Librascope Division
808 Western Ave.
Glendale 1, California

GENERAL PRECISION INC.
Kearfott Div.
1150 McBride Ave.
Little Falls, New Jersey

H-W ELECTRONICS INC.
14 Huron Drive
Natick, Massachusetts

HUGHES AIRCRAFT CO.
Fullerton R&D
P.O. Box 2097
Fullerton, California

INTERNATIONAL TELEPHONE & TELEGRAPH CORP.
ITT Federal Laboratories
500 Washington Ave.
Nutley 10, New Jersey

LITTON INDUSTRIES, INC.
Guidance and Control Systems Div.
5600 Canoga Ave.
Woodland Hills, California

MATHATRONICS, INC.
621 Main Street
Waltham 54, Massachusetts

NATIONAL CASH REGISTER CO.
Main and K Streets
Dayton 9, Ohio

NORTHROP CORP.
Nortronics Division
Research Park
Palos Verdes, California

PACIFIC DATA SYSTEMS, INC.
1058 E. First Street
Santa Ana, California

PHILCO CORP.
Subsidiary of Ford Motor Co.
Computer Division
3900 Welsh Road
Willow Grove, Pennsylvania

PHILCO CORP.
Subsidiary of Ford Motor Co.
4700 Wissahickon Ave.
Philadelphia 44, Pennsylvania

SCHUTTER MICROWAVE CORP.
80 E. Montauk Hwy.
Linpenhurst, Long Island, New York

SCIENTIFIC DATA SYSTEMS INC.
1542 15th Street
Santa Monica, California

SPERRY RAND CORP.
Ford Instruments Co.
31-10 Thomson Ave.
Long Island City 1, New York

TAMAR ELECTRONICS INC.
American Gyro Div.
9320 Lincoln Blvd.
Los Angeles 45, California

TELECOMPUTING CORP.
9229 Sunset Blvd.
Los Angeles 69, California

TOMPSON RAMO WOOLDRIDGE INC.
8433 Fallbrook Ave.
Canoga Park, California

UNITED AERO SPACE
200 Allendale Rd.
Pasadena, California

PART B

ACF ELECTRONICS
Division of ACF Industries
Lafayette Street
Riverdale, Maryland

ABACUS INC.
1718 21st Street
Santa Monica, California

A. C. SPARK PLUG DIV.
General Motors Co.
7929 S. Howell Ave.
Milwaukee 1, Wisconsin

AD-VANCE R/D. CO.
2930 San Luis Rey Road
Oceanside, California

AEROJET GENERAL CORP.
P. O. Box 296
Azusa, California

AERONCA MANUFACTURING CORP.
Aerospace Division
P.O. Box 536
Baltimore 3, Maryland

AIRCRAFT ARMAMENTS, INC.
Cockeysville, Maryland

AMERICAN OPTICAL CO.
J. W. Fecker Division
4709 Baum Blvd.
Pittsburgh 13, Pennsylvania

APPLIED DEVELOPMENT CORP.
1131 Monterey Pass Rd.
Monterey Park, California

BENDIX CORP.
Bendix-Pacific Division
11600 Sherman Way
North Hollywood, California

BOOZ ALLEN APPLIED RESEARCH, INC.
Designers for Industry
4241 Fulton Pkwy.
Cleveland 9, Ohio

CONSOLIDATED MICROWAVE CORP.
850 Shepherd Ave.
Brooklyn 8, New York

CRAIG SYSTEMS INC.
360 Merrimack Street
Lawrence, Massachusetts

DELCO RADIO DIV.
General Motors Corp.
700 E. Firmin Street
Kokomo, Indiana

DEWEY CORP.
202 E 44th Street
New York 17, New York

DORNE AND MARGOLIN, INC.
29 New York Ave.
Westbury, Long Island, New York

ELECTRA SCIENTIFIC CORP
Electra Way
Fullerton, California

ELECTRONIC SPECIALITY CO.
5121 San Fernando Road
Los Angeles 39, California

EMERTRON, INC.
1140 East-West Hwy.
Silver Springs, Maryland

GENERAL DATA CORP.
1250 N. Parker
Orange, California

GENERAL DYNAMICS/POMONA
Box 1011
Pomona, California

GENERAL ELECTRIC CO.
Light Military Electronics
French Road
Utica, New York

GENERAL MILLS
Electronics Div.
1620 Central Ave.
Minneapolis 18, Minnesota

GENERAL PRECISION INC.
50 Prospect Ave.
Tarrytown, New York

GENERAL MICROWAVE CORP.
155 Marine Street
Farmingdale, Long Island,
New York

GEOTECHNICAL CORP.
3401 Shiloh Road
Garland, Texas

GILFILLAN CORP.
1815 Venice Blvd.
Los Angeles 6, California

GORDON ENTERPRISES
5362 Cahuenga Blvd.
North Hollywood, California

GORHAM ELECTRONICS DIV.
Gorham Corp.
333 Adelaide Ave.
Providence 7, Rhode Island

HARMON-KARDON, INC.
Ames Ct.
Plainview, Long Island, New York

HUYCK SYSTEMS CO., DIV
HUYCK CORP.
360 Wolf Hill Road
Huntington Station, Long Island,
New York

INSTRUMENTS FOR INDUSTRY
INC.
101 New South Road
Hicksville, New York

INTERCONTINENTAL
ELECTRONICS CORP.
Affil. of CSF
300 Shames Drive
Westbury, Long Island, New York

KILLMORGAN CORP.
Inland Motor Corp.
347 King Street
Northampton, Massachusettes

LING-TEMCO-VOUGHT, INC.
Temco Electronics Div.
Temco Electronics and Missile Co.
P.O. Box 6118
Dallas 22, Texas

MAGNETIC RESEARCH CORP.
3160 W. El Segundo Blvd.
Hawthorne, California

MARTIN CO.
Martin-Marietta Corp.
Friendship International Airport
Baltimore 40, Maryland

MASER OPTICS, INC.
89 Brighton Ave.
Boston 34, Massachusetts

MICRO GEE PRODUCTS INC.
6319 W. Slauson Ave.
Culver City, California

MODEL ENGINEERING AND MFG. CORP.
Huntington, Indiana

MOTOROLA INC.
Military Electronics Div.
8201 E. McDowell Road
Scottsdale, Arizona

NORBATROL ELECTRONICS CORP.
356 Collins Ave.
Pittsburgh 6, Pennsylvania

PACKARD BELL
1905 Armacost Ave,
Los Angeles, California

POLARAD ELECTRONICS CORP.
43-20 34th Street
Long Island City 1, New York

RS ELECTRONICS GROUP
795 Kifer Road
Sunnyvale, California

RADIATION, INC.
Melbourne, Florida

RANSOM RESEARCH, DIV WYLE LABS
Box 269 374 W. 8th St.
San Pedro, California

RESE ENGINEERING, INC.
A and Courtland Streets
Philadelphia 20, Pennsylvania

SANDERS ASSOCIATES INC.
95 Canal Street
Nashua, New Hampshire

SCOPE INC.
121 Fallfax Drive
Falls Church, Virginia

SERVOMECHANISMS/INC.
200 N. Aviation Blvd.
El Segundo, California

SIER-BATH GEAR CO.
9252 Hudson Blvd.
North Bergen, New Jersey

SIERRA RESEARCH CORP.
P. O. Box 22
Buffalo 25, New York

SOLID STATE ELECTRONIC CO.
15321 Rayen St.
Sepulveda, California

SPACEONICS, INC.
2 Overhill Road
Scarsdale, New York

STEWARD-WARNER ELECTRONICS DIV.
Stewart-Warner Corp.
1300 N. Kostner Ave.
Chicago 51, Illinois

SUMMIT INDUSTRIES INC.
1515 S. Santa Fe
Compton, California

SYLVANIA ELECTRIC PRODUCTS, INC.
Sylvania Electronic Systems Division
730 Third Avenue
New York 17, New York

SYSTEMS DESIGN INC.
7604 San Fernando Road
Sun Valley, California

TAMAR ELECTRONICS INC.
Tamar Electronics Div.
2045 Rosecrans Blvd.
Gardena, California

TEXTRON INC.
Dalmo Victor Co.
1515 Industrial Way
Belmont, California

WATERMAN PRODUCTS CO., INC.
2445 Emerald Street
Philadelphia 25, Pennslyvania

WATKINS-JOHNSON CO.
3333 Hillview Avenue
Palo Alto, California

WAYNE GEORGE CORP.
Dunn Engineering Div.
322 Needham Street
Newton 64, Massachusetts

WILCOX ELECTRIC CO., INC.
1400 Chestnut Street
Kansas City 27, Missouri

PART C

ADMIRAL CORP.
3800 W. Cortland
Chicago 47, Illinois

ADVANCED DEVELOPMENT LABORATORIES, INC.
Haines Street
Nashua, New Hampshire

AEROFLEX LABORATORIES INC.
48-25 36th Street
Long Island City 1, New York

AEROJET-GENERAL CORP.
Space General Corp.
9200 E. Flair Drive
El Monte, California

ALLISON CONSULTING ENGINEERS
3110 Edith Blvd. N. E.
Albuquerque, New Mexico

AMERICAN BOSCH ARMA CORP.
Arma Div.
Roosevelt Field
Garden City, Long Island, New York

AMERICAN BOSCH ARMA CORP.
Roosevelt Field
Garden City, Long Island, New York

AMICON CORP.
5903 City Ave.
Philadelphia 31, Pennsylvania

AVCO CORP.
Electronics and Ordnance Div.
Electronics Operation
Cincinnati 41, Ohio

BELL AEROSYSTEMS CO., DIV.
Bell Aerospace Corp.
Textron Co.
P.O. Box 1
Buffalo 5, New York

BELOCK INSTRUMENT CORP.
111-01 14th Avenue
College Point 56, New York

BENDIX CORP.
Bendix Computer Div.
5630 Arbor Vitae Street
Los Angeles 45, California

BENDIX CORP.
Eclipse Pioneer Div.
Teterboro, New Jersey

BROWN ENGINEERING CO., INC.
P. O. Drawer 917
Huntsville, Alabama

CBC ELECTRONICS CO., INC.
2601 N. Howell Street
Philadelphia 33, Pennsylvania

CHURCHILL LIGHTING CORP.
344 Franklin Street
Melrose 76, Massachusetts

CLARY CORP.
408 Junipero Street
San Gabriel, California

COLLINS RADIO CO.
Dallas Div
Dallas, Texas

COOK ELECTRIC CO.
NRK Microwave Div.
4601 W. Addison Street
Chicago 41, Illinois

COOK TECHNOLOGICAL CENTER DIV.
Cook Electric Co.
6401 Oakton Street
Morton Grove, Illinois

CUTLER-HAMMER
Airborne Instruments Laboratory Div.
Comac Road
Deer Park, Long Island, New York

DAYSTROM, INC.
430 Mountain Ave.
Murray Hill, New Jersey

DAYTON ELECTRONIC PRODUCTS CO. INC.
915 Webster Street
Dayton 4, Ohio

DECITRON ELECTRONICS CORP.
850 Shepherd Ave.
Brooklyn 8, New York

DEFENSE ELECTRONICS INC.
5455 Randolph Road
Rockville, Maryland

DIGITAL DEVELOPMENT CORP.
7541 Eads Avenue
La Jolla, California

DJECO DIV.
Djordjevic Engineering Corp.
1933 N. Damen
Chicago 47, Illinois

DOUGLAS RESEARCH CORP.
Div. Douglas Microwave Corp.
254 E. Third Street
Mount Vernon, New York

E-Z WAY TOWERS INC.
5901 E. Broadway
P. O. Box 5767
Tampa 5, Florida

ELECTRO CIRCUITS, INC.
176 Walker Street
Lowell, Massachusetts

ELECTRONIC PRODUCTS CORP.
2315 Cecil Ave.
Baltimore 18, Maryland

ENGINEERING MODEL LABORATORY, INC.
65 Union Street
Ashland, Massachusetts

FAIRCHILD CAMERA AND INSTRUMENT CORP.
Defense Products Div.
300 Robbins Lane
Syosset, Long Island, New York

FAIRCHILD STRATOS CORP.
Hagerstown 10, Maryland

FIDELITONE MICROWAVE INC.
JVM Div.
9300 W. 47th Street
Brookfield, Illinois

FRANKLIN CO., INC.
4045 Torresdale Ave.
Philadelphia 24, Pennsylvania

GENERAL APPLIED SCIENCE LABORATORIES, INC.
Merrick and Stewart Avenues
Westbury, Long Island, New York

GENERAL DYNAMICS/
ELECTRONICS
P. O. Box 127
San Diego 12, California

GENERAL ELECTRONIC CO.
Radar
Light Military Electronics Dept.

GENERAL ELECTRIC CO.
Ordnance Dept.
100 Plastics Avenue
Pittsfield, Massachusetts

GENERAL INSTRUMENT CORP.
Defense and Engineering Products Group
Andrews Road
Hicksville, Long Island, New York

GENERAL PRECISION INC.
Kearfott Semiconductor Corp.
437 Cherr Street
West Newton 65, Massachusetts

GENERAL RAILWAY SIGNAL CO.
Cardion Electronics, Inc.
65 Rushmore Street
Westbury, Long Island, New York

GOODYEAR AIRCRAFT CORP.
1210 Massillon Road
Akron 15, Ohio

GOSSLAND ENGINEERING CO.
848 New York Drive
Altadena, California

HALLICRAFTERS CO.
Fifth and Kostner Avenues
Chicago 24, Illinois

HAZELTINE CORP.
59-25 Little Neck Pkwy
Little Neck 62, New York

HUGHES INDUSTRIES
Riverton, New Jersey

INFORMATION SYSTEMS
10131 National Blvd.
Los Angeles 34, California

INTERNATIONAL BUSINESS
MACHINES CORP.
590 Madison Avenue
New York 22, New York

INTERNATIONAL BUSINESS
MACHINES CORP.
Federal Systems Div.
326 Montgomery Avenue
Rockville, Maryland

IRVING AIR CHUTE CO., INC.
Suite 901
565 Fifth Avenue
New York 17, New York

KAMAN AIRCRAFT CORP.
Old Windsor Road
Bloomfield, Connecticut

LABORATORY FOR ELECTRONICS
LFE Electronics Div.
985 Commonwealth Avenue
Boston 15, Massachusetts

LEAR-SIEGLER, INC.
Data and Control Division
34-01 38th Avenue
Long Island City 1, New York

LEAR-SIEGLER, INC.
Instrument Division
110 Ionia Avenue N. W.
Grand Rapids, Michigan

LEAR-SIEGLER, INC.
Data and Controls Div.

LING-TEMCO-VOUGHT, INC.
Panellit Division
ISI, Inc.
7401 N. Hamlin Avenue
Skokie, Illinois

LING-TEMCO-VOUGHT, INC.
P. O. Box 5003
Dallas 22, Texas

LING-TEMCO-VOUGHT, INC.
Temco Aero Systems Division
Temco Electronics and Missiles Co.
P. O. Box 1056
Greenville, Texas

LIONEL CORP.
Telerad Division
Rt. 69
Flemington, New Jersey

LITTON INDUSTRIES
336 N. Foothill Road
Beverly Hills, California

LITTON INDUSTRIES, INC.
Monroe Calculating Machine Co.
Electronics Component Div.
60 Main Street
San Francisco 5, California

LOCKHEED ELECTRONICS CO.
Div. of Lockeed Aircraft Corp.
U. S. Hwy 22
Plainfield, New Jersey

LOCKHEED MISSILES AND SPACE CO.
P. O. Box 504
Sunnyvale, California

MAGNAVOX CO.
Magnavox Research Laboratories
2829 Maricopa Street
Torrance, California

MATRIX RESEARCH AND DEVELOPMENT CORP.
11 Mulberry Street
Nashua, New Hampshire

MAXSON ELECTRONICS CORP.
475 Tenth Avenue
New York 18, New York

MELABS
3300 Hillview Avenue
Palo Alto, California

MICROWAVE ASSOCIATES, INC.
Northwest Industrial Park
Burlington, Massachusetts

MINNEAPOLIS-HONEYWELL REGULATOR CO.
Electronic Data Processing Div.
60 Walnut Street
Wellesley Hills 81, Massachusetts

MINEAPOLIS-HONEYWELL REGULATOR CO.
Military Products Group
2755 Fourth Avenue S.
Minneapolis 8, Minnesota

NORTH ELECTRIC CO.
Galion, Ohio

NORTHERN INSTRUMENT CORP.
77-79 E. Main Street
Bay Shore, Long Island, New York

OTIS ELEVATOR CO.
Defense and Industrial Div.
35 Ryerson Street
Brooklyn 5, New York

PACIFIC STATES ENGINEERING
12838 Weber Way
Hawthorne, California

PARSONS ELECTRONICS DIV.
Ralph M. Parsons Co.
157 S. DeLacey Avenue
Pasadena, California

PENNWOOD NUMECHRON CO.
7249 Frankstown Avenue
Pittsburgh 8, Pennsylvania

PLUG-IN INSTRUMENTS, INC.
1416 Lebanon Road
Nashville 10, Tennessee

POWERTRONICS SYSTEMS INC.
10-12 Pine Court
New Rochelle, New York

RHG ELECTRONICS
LABORATORY, INC.
94 Milbar Blvd.
Farmingdale, Long Island,
New York

RLC ELECTRONICS, INC.
25 Martin Place
Port Chester, New York

RADIATRONICS, INC.
14801 Califa Street
Van Nuys, California

RADIO CORPORATION OF
AMERICA
Defense Electronic Products
Front and Cooper Streets
Camden 2, New Jersey

RAY, DAISLEY CO.
585 W. Hoffman Avenue
Lindenhurst, Long Island,
New York

RAYTHEON CO.
Government Marketing
Lexington 73, Massachusetts

REEVES INSTRUMENT CO.
Roosevelt Field
Garden City, Long Island,
New York

RELIANCE ELECTRIC AND
ENGINEERING CO.
24701 Euclid Avenue
Cleveland 17, Ohio

RYAN AERONAUTICAL CO.
Lindbergh Field
San Diego 12, California

SIEMENS NEW YORK. INC.
350 Fifth Avenue
New York 1, New York

SOMERSET RADIATION
LABORATORY. INC.
192 Central Avenue
Stirling, New Jersey

SPERRY GYROSCOPE CO.
Marcus Avenue.
Great Neck, Long Island,
New York

SPERRY RAND CORP.
Sperry Phoenix Co.
21111 N. 19th Avenue
Phoenix 27, Arizona

SPERRY RAND CORP.
Sunnyvale Development Center
Sperry Phoenix Co.
294 Commercial Street
Sunnyvale, California

SPERRY RAND CORP.
Univac Military Operations
Univac Park
Saint Paul 16, Minnesota

SPERRY RAND CORP.
Wheeler Electronics Co.
150 E. Aurora Street
Waterbury 20, Connecticut

STRAND ENGINEERING CO.
P. O. Box 76
Ann Arbor, Michigan

TRG INC.
2 Aerial Way
Syosset, Long Island, New York

TAFFET ELECTRONICS, INC.
27-01 Brooklyn-Queens Expwy. W.
Woodside 77, New York

TARC ELECTRONICS DIV.
Gotham Broadcasting Corp.
48 Urban Avenue
Westbury, Long Island, New York

TECH SERV, INC.
4911 College Avenue
College Park, Maryland

TECHNITROL, INC.
1952 E. Allegheny Avenue
Philadelphia 34, Pennsylvania

TELECTRO INDUSTRIAL CORP.
35-16 37th Street
Long Island City 1, New York

TELFREGISTER CORP.
445 Fairfield Avenue
Stamford, Connecticut

TEN BOSCH INC.
80 Wheeler Avenue
Pleasantville, New York

TEXAS INSTRUMENT INC.
Apparatus Div.
P. O. Box 6015
Dallas 22, Texas

TRANSDYNE CORP.
43 Albertson Avenue
Albertson, Long Island, New York

UNISTRUT PRODUCTS CO.
933 W. Washington
Chicago 7, Illinois

UNITED AIRCRAFT CORP.
Norden Div.
Helen Street
Norwalk, Connecticut

VARIAN ASSOCIATES
611 Hansen Way
Palo Alto 2, California

VEEDER-ROOT, INC.
70 Sargeant Street
Hartford 2, Connecticut

WAVETRONICS INC.
324 W. Westfield Ave.
Roselle Park, New Jersey

WESTINGHOUSE ELECTRIC CORP.
Gateway Center
Pittsburgh 30, Pennsylvania

WESTINGHOUSE ELECTRIC CORP.
Air Arm Div.
P. O. Box 746
Baltimore 3, Maryland

WINDER AIRCRAFT CORP. OF
FLA.
P. O. Drawer No. 8
Dunnellon, Florida

APPENDIX VI. NOMENCLATURE

Symbols	Definitions	Units
X, Y	Coordinates of runway takeoff end computed from sensor information.	Ft
X_I, Y_I, Z_I	Inertial coordinate system axes.	Ft
X_i, Y_i, Z_i	Airport coordinate system axes based at runway center.	Ft
X_A, Y_A, Z_A	Airplane coordinates system axes.	Ft
x, y, z	Airframe body axis coordinate system axes.	Ft
$\dot{X}_h$, $\dot{Y}_h$, $\dot{Z}_h$	Aircraft inertial rates.	Ft/sec
x_d, y_d	Display coordinate system axes.	In

Symbols	Definitions	Units
$\left.\begin{matrix} y_D \\ z_D \end{matrix}\right\}$	Display coordinate system axes.	In.
$\left.\begin{matrix} \bar{I} \\ \bar{J} \\ \bar{K} \end{matrix}\right\}$	Unit vectors in airport coordinate system.	
$\left.\begin{matrix} \bar{i} \\ \bar{j} \\ \bar{k} \end{matrix}\right\}$	Unit vectors in airframe body axis coordinate system.	
$(\ )_A$	Function in airplane coordinates, or function for altitude marker.	
$(\ )_i$	Function in airport coordinates.	
$(\ )_V$	Function for velocity circle.	
$(\ )_H$	Function for horizon line.	
$(\ )$	Rate of change of a variable.	
$\Delta(\ )$	Increment of a variable.	
R	Radar slant range to runway takeoff end.	Ft
R_ℓ	Range along flight path.	
$\left.\begin{matrix} R_a \\ R_b \\ R_c \\ R_d \end{matrix}\right\}$	Slant range to appropriate DME sensor.	Ft
R_A	Slant range to origin of airport coordinate system.	Ft
h	Altitude above runway	Ft

Symbols	Definitions	Units
h_o	Altitude marker maximum indicated altitude.	Ft
h_1	Barometric altitude from altimeter one.	Ft
h_2	Barometric altitude from altimeter two.	Ft
h_r	Radar altitude.	Ft
h_t	Transponder height above runway.	Ft
h_B	Barometric altitude.	Ft
a	Distance from runway takeoff end to runway intermediate line, 5000 ft.	Ft
ℓ	(APPA, C) One-half of beacon separation distance in X runway direction.	Ft
ℓ	(APPB) Length of runway, 10,000 ft.	Ft
m	One-half of beacon separation distance in Y runway direction.	Ft
n	Height of fourth beacon above other three	Ft
b	Width of runway, 250 ft.	Ft
$V_{W/E}$	Wind velocity from East.	Ft/sec
$V_{W/S}$	Wind velocity from South.	Ft/sec
K_θ	Angular-to-linear conversion scale factor	In/ radian
ψ_0	Runway heading angle measured from North	Deg, radians
ψ	Yaw or aircraft heading angle measured from North	Deg, radians
ψ	(APPC) used as ground track angle	Deg

Symbols	Definitions	Units
A through L	Display computation parameters defined elsewhere.	In
R_h, R'_h, R''_h	Range computation parameters defined elsewhere.	Ft
ξ	Radar antenna azimuth angle.	Deg
ξh	Heading angle to runway takeoff end.	Deg
η	Radar antenna elevation angle.	Deg
ρ	Velocity circle radius.	In
ϕ	Roll Angle.	Deg, radians
θ	Pitch Angle.	Deg, radians
γ	Longitudinal flight path angle.	Deg
α	Angle of attack.	Deg
α_0	Aircraft static angle of attack.	Deg
P	Roll rate.	Deg/ sec
V_o	Velocity along flight path.	Ft/ sec
C_L	Lift coefficient.	
C_{L_o}	Lift coefficient neglecting ground efficient.	
C_M	Pitching moment coefficient.	

Symbols	Definitions	Units
C_{M_o}	Pitching moment coefficient neglecting ground effect.	
C_D	Drag coefficient.	
δ_{S_E}	Stick deflection in pitch.	In
δ_{S_A}	Stick deflection in roll.	In
K_{TRIM}	Trim button gain.	Deg/ sec
δ_{TRIM}	Discrete trim button position.	
δ_E	Elevator Deflection.	Deg
δ_A	Aileron deflection.	Deg
K_{δ_E}	Gain from stick to elevator.	Deg/ in
K_{δ_A}	Gain from stick to ailerons.	Deg/ in
s	Laplacian operator.	1/radian

APPENDIX VII. GENERAL ENVIRONMENTAL AND EQUIPMENT SPECIFICATIONS

A. GENERAL SPECIFICATIONS

The following general specifications are recommended as requirements, where applicable, for the various equipment to be used in the system.

1. Materials and Parts

Materials and parts used shall be military standard material and parts, or equivalent. The use of a non-standard material or part is acceptable in the absence of a standard part, or if its use will result in a substantial improvement in performance, reliability, simplicity of design or operation.

a. Materials

The materials, component parts, and mechanical assemblies used shall be of the lightest practical weight, and of quality compatible with the specified performance. Nonflammable material and weight-saving designs shall be used whenever possible. Materials and component parts that have fungus nutrient characteristics shall not be used. General materials specifications are as follows:

1. Toxic Material. Materials when in storage or normal operation shall not produce dangerous gases or other harmful toxic effects.

2. Metals. Corrosion resistant metals shall be used whenever possible.

3. Dissimilar Metals. Dissimilar metals shall not be used in intimate contact, unless suitably protected against electrolytic corrosion.

4. Mercury. The use of mercury or any mercury compound is specifically prohibited.

5. Aluminum. Aluminum alloys may be used where a saving in weight will result, provided that all performance and test requirements are met.

6. Wood. Wood (treated or untreated) shall not be used.

7. Plastics. The use of plastic material is acceptable for the support of printed circuit boards and connectors. The use of plastic material for the construction of printed circuit boards is permissable.

8. Thermoplastic Materials. The use of thermoplastic materials shall be limited to those which meet the temperature requirements.

9. Protective Treatment. When parts and materials are used that are subject to deterioration when exposed to the expected climatic and environmental conditions, they shall be protected against such deterioration in a manner that will in no way prevent compliance with the performance requirements.

b. Parts - Mechanical

The following specifications apply to mechanical parts:

1. Hardware. Stainless steel fasteners (bolts, screws, nuts, cotter pins, washers, etc) shall be utilized whenever possible.

2. Ball Bearings. Ball bearings, if used, shall be in accordance with MIL-I-983. Precision grade 50 bearings may be used.

c. Parts - Electrical

The following specifications apply to electrical parts:

1. Electron Tubes. Electron tubes shall not be used unless specifically approved by the procuring activity.

2. Capacitors. Electrolytic capacitors are permitted. Tantalum electrolytic capacitors conforming to specification MIL-C-3965 are preferred.

3. Transformers. Transformers shall be grade 5, class S and life expectancy "X" in accordance with specification MIL-T-27.

4. Semiconductor Parts. Every effort shall be made to restrict the use of transistors and diodes to the smallest practical number of different types.

d. Printed Wiring Boards

The following specifications shall apply to printed wiring boards:

1. Eyelets. Eyelets shall not be used on printed wiring boards for electrical contact. All parts leads shall be soldered directly to the printed circuit material for electrical contact.

2. Clinching of Part Leads. Part leads shall not be clinched, but may be bent sufficiently to aid in the soldering.

3. Mounting of Electronic Parts. Electronic components may be mounted on both sides of printed circuit boards. Jumper wires may be used to interconnect circuitry where necessary. The clearance between the soldered connection and the body of the part, when measured along the part lead, shall not be less than 1/8 in. The total length of both leads from the soldered connection to the body of the parts shall not exceed 1 in. Provision for stress relief must exist on all component leads after soldering.

4. Functional Test Points. Functional test points shall be provided on printed circuit boards as necessary to facilitate board tests.

5. Hookup Wire and Cable. Numbering of wiring is not required.

6. Connectors and Receptacles. Connectors shall be per MIL-C-26482 (ASG). Identical types of receptacles shall have their guide key rotated with respect to each other. Receptacles used with printed circuit boards shall have the guide pins to correspond with the keying slot in the mating printed circuit board.

2. General Design Requirements

Equipment shall be designed primarily for maximum functional capability and reliability, but minimum space and weight consistent with solid-state devices design techniques shall also be considerations. The following general design requirements shall apply:

1. Fire Hazard. The design and construction of the equipment shall be such that its use shall not, under conditions of storage or operation, create a fire hazard.

2. Interchangeability. Serialized assemblies having the same manufacturer's part number shall be functionally and dimensionally interchangeable.

3. Identification and Serialization of Subassemblies. Printed circuit boards and all subassemblies attached by marked connectors for easy removal shall be identified with the following information:

   a. Manufacturer's part number
   b. Manufacturer's name/code name
   c. Serial number

   When nameplates are used to identify subassemblies, identification and designation plates for units and subassemblies shall be in accordance with specification MIL-P-15024. Serial numbers on identification plates are required.

4. Electrical Circuit Construction. Electrical circuits shall be constructed on plug-in modules. Replacement of any electronic module or mechanical subassembly shall require no longer than 5-min time. Identical plug-in units shall be used where possible to minimize the spare parts requirements.

5. Human Engineering. The equipment shall be designed to ensure maximum efficiency, ease of operation, operator safety, and minimum degradation and variability in man-machine system performance as applicable. The following are some of the human engineering factors which shall be considered:

   a. Space limitations for operation and maintenance.
   b. Environmental factors during operation and maintenance.
   c. Level of training and psychophysical limiation on operators.
   d. Human safety factors.

6. Wormanship. The equipment, including all parts and accessories, shall be constructed and finished in a thoroughly workmanlike manner. Particular attention shall be paid to neatness of assembly, ease of manufacture, and quality control. Attention shall be given to freedom from defects, burrs and sharp edges; thoroughness of soldering, welding, painting, wiring, and riveting; alinement of parts; and tightness of assembly.

3. <u>Environment</u>

The following environmental specifications shall apply:

1. Pressure - Operating: The equipment shall operate without malfunction in an ambient pressure environment of 14.74 ± 3 psi.

The equipment shall withstand pressures between 10 and 30 psi, changing at a maximum rate of 3 psi per second without permanent degradation.

2. Temperature - Operating: 40 to 130 F ambient air. Non-operating: -20 to +160 F ambient. Cooling air may be used.

3. Humidity - Operating: The equipment shall suffer no damage, failure or malfunction when subjected to Procedure I of MIL-E-5272C with the following modifications: Temperature 40 to 130 F, relative humidity 80 percent, 5 cycles (120 hr). Non-operating: Same as operating, except that the equipment may be dried with forced air for 12-hr following the 5th cycle before power is applied.

4. Acceleration - Operating: 2 g in any direction. Nonoperating: Same as operating.

5. Vibration - Operating: 2 g, 5 to 40 cps in accordance with MIL-STD-167. Nonoperating: 3 g, 10 to 200 cps in accordance with MIL-STD-167.

6. Shock - The equipment shall not suffer permanent damage or performance degradation after being subjected to a shock of 10 g in any direction. Nonoperating: Same as operating.

7. Acoustic Noise - Operating: The equipment shall operate without malfunction when exposed to an ambient acoustic noise level between 20 and 2,000 cps, of 70 db. The equipment shall withstand 300 and 4,800 cps, of 120 db for 2 sec without permanent degradation. Nonoperating: Same as operating.

4. Conducted and Radiated Susceptibility

The equipment shall be designed to minimize susceptibility to interference from other sources. The enclosing case construction shall be designed not only to minimize interference propagation, but also to minimize interference pickup from external sources. Where conducted engergy on the power leads or any external leads might cause interference, the leads shall be isolated from other leads to avoid coupling, and where necessary shall have line filters at their entry into the enclosing case. No change in indication, or degradation of performance below that specified, shall be produced in the equipment when subjected to the following signals:

1. Conducted (RF) Signal. Match load, swept single-frequency signals from a 14 kc to 1000 mc, 100,000 uv source having an impedance of 50 ohms applied at the input leads of the equipment, including power input leads by means of appropriate line stabilization networks.

2. Radiated (RF) Signal. 100,000-uv signals from a 14 kc to 1000 mc radiating source as follows:

Frequency	Antenna
14 to 150 kc	5 in. loop
150 to 20 kc	41 in. rod
20 to 1000 kc	Tune dipole

   The antenna shall be located 3 ft from the test sample, oriented in the maximum field position in each of two, 90 deg apart, planes.

3. Audiofrequency Noise Reduction. Interference radiated from the equipment or interconnecting wiring between the equipment and associated periphery equipment shall not exceed 100 uv per meter per kc (peak) over the frequency range from 30 cps to 14 kc.

4. Radiated Interference. Interference radiated from the equipment interconnecting wiring between the equipment and associated periphery equipment shall not exceed 100 uv per meter per kc (peak) over the frequency range from 30 cps to 14 kc.

5. Conducted Interference. Interference voltage appearing on data transmission lines between the equipment shall not exceed 200 uv/kc bandwidth (peak) over the frequency range from 30 cps to 14 kc.

6. Conducted Susceptibility. The equipment shall be designed to operate within the performance requirements of this specification when a 3-v, 50 cps to 14 kc, signal is superimposed on the primary power lines. It shall also be designed to operate within

the performance requirements of this specification when a 3 mv, 50 cps to 14 kc, signal is superimposed on the input signal leads. Input signal leads are defined as leads which conduct intelligence from associated peripheral equipment.

7. Radiated Susceptibility. The equipment shall be designed to operate within the performance requirements of this specification when subjected to radiated signals, 50 cps to 14 kc, from an antenna with an antenna current of 1 amp. The antenna current shall be supplied to a 10-turn loop antenna, 30 in. in diameter. The antenna should be moved about the equipment maintaining a distance of at least 1 ft from the equipment.

B. DIGITAL COMPUTER SPECIFICATIONS

The following are recommended as requirements for a blind landing system digital computer. The specifications for physical characteristics, life, reliability and maintainability requirements, electrical characteristics, functional requirements, environmental requirements, control panel requirements, and software requirements follow.

1. Physical Characteristics

The physical characteristics of the computer shall be as follows:

1. Configuration and Size. The maximum volume shall be less than 4 cu ft. The maximum dimension shall be less than 36 in.

2. Weight. The weight of the computer shall not exceed 200 lb.

3. Mounting. The computer shall meet the functional requirements of this specification when mounted in, or inclined to any, position.

4. Identification. The computer shall be supplied with a nameplate in accordance with MIL-P-15024.

5. Packaging. The memory and all printed circuit boards shall be accessible and removable when the environmental covers are removed.

2. Life, Reliability, and Maintainability Requirements

The life, reliability and maintainability requirements shall be as follows:

1. Operating Life. The computer shall be designed for a minimum life expectancy of 2,000 hr when operating under the various combinations of environment encountered during operational conditions and when subject to overhauls, occurring at intervals of not less than 600 operating hours, or 4,000 total hours, whichever occurs first.

2. Storage. The computer shall be designed to withstand without permanent degradation, a storage period of 5 years in an environment as described.

3. Reliability. The reliability goal under operating environments, shall be a mean time between failure of at least 500 hr.

4. Maintainability. The equipment shall be designed to provide for ease of maintenance in the operating environments. The maintainability goal expressed as an availability ratio shall be 0.985. Sufficient information should be supplied to allow for maintenance by technicians available to the procuring agency.

5. Lubrication. Lubrication during normal maintenance operations shall not be required.

6. Special Tools. The requirements for special tools shall be held to a minimum. Special tools shall be designed and constructed to withstand normal abuse throughout the life of the computer.

7. Repairability. The repairability design goal for the computer shall be a mean time to accomplish repair of a maximum of 8 hr. Computer maintenance and repair shall be designed to be carried out by replacing subassemblies (module boards). However, technicians shall be supplied with sufficient information to isolate failures to particular module boards, and to repair these module boards. Electrical adjustments shall be held to a minimum.

3. Electrical Characteristics

The electrical characteristics shall be as follows:

1. Main Power. Main power shall be supplied by a nominal 120-v line-to-neutral, 3-phase 3-2i43, 400-cps static inverter, or an equivalent device unless otherwise indicated in this specification. Tape shall be included on the main power transformer in the computer to allow excitation to be obtained from a 200-v, or a 115-v line-to-neutral 3-phase, 400-cps source. All main power requirements of the computer shall be obtained from secondary windings of the power transformer. Main power inputs shall normally be isolated from all grounds within the computer. An electrostatic shield shall be included between the primary and secondaries of the transformer. The shield shall be connected internally to the computer power ground. Electrical characteristics are specified on the basis of a 120-v input voltage. These characteristics must be changed accordingly for 115-v or 200 v inputs.

2. Main Power Line-to-Line Steady-State Voltage. The computer shall be capable of operating satisfactorily when the main power line-to-neutral steady-state voltage is within the following limits:

   a. Upper Limit 126.0 v rms
   b. Lower Limit 114.0 v rms

3. Main Power Steady-State Voltage Unbalance. The deviation from the average voltage as measured from phase to phase shall be less than 5 percent. Transients outside those limits shall cause the computer to enter a program protection mode. The disturbances shall not occur at intervals less than 10 sec.

4. Main Power Voltage Waveform. The main power waveform shall be within the following limits with the computer operating:

   a. Crest factor: 1.41 ± 0.14
   b. Total harmonic content: 5 percent of the fundamental (rms) when measured with a distortion meter as distortion of the fundamental frequency.

5. Main Power Frequency. The computer shall operate satisfactorily when the main power frequency is within the range of 380 to 420 cps.

6. Main Power Amplitude Modulation. The computer shall operate satisfactorily when the amplitude of the main power is varying 3.2 v. The measurement shall be the peak-to-valley difference between the minimum voltage reached and the maximum voltage reached on the modulation envelope over a periof of at least 1 sec.

7. High Voltage Transients. The computer shall be capable of operating satisfactorily and without entering a program protection mode when transient voltages on the main power line increase the line-to-line voltage to 375 v peak-to-peak for a time duration of 76 usec or less. The computer shall be capable of withstanding without detrimental or adverse effect to the circuitry within the computer peak-to-peak (line-to-line) transient voltages of 405 v for a duration of 150 usec.

8. Low Voltage Transients and Power Interruptions. The computer shall be capable of operating satisfactorily and without entering the program protection mode when voltage disturbances on the main power reduce the line-to-line voltage to 300 v peak-to-peak for a time duration of 76 usec or less, or power is interrupted for periods of 76 usec or less.

9. Internal Overload Protection. No circuit breakers or fuses shall be utilized within the computer.

4. Computer Functional Requirements

The computer functional requirements shall be as follows:

1. Type. The computer shall have, as a minimum, a whole number arithmetic center and an input/output center capable of solving the recommended equations given in Appendix I, sampling the inputs and generating the outputs indicated.

2. Control and Data Processing. The computer shall be controlled by its own action from an internally-stored program entered into the memory from an external device.

3. Word Length. The word length shall be a minimum of 20 bits plus sign.

4. Word Format. A word may have one of two formats, depending on its use as an instruction or a full number.

5. Instructions. The computer shall have normal arithmetic operations required to solve the equations as given in Appendix I.

6. Arithmetic Speed. The computer shall have the capability of providing complete solutions to the equations in Appendix I with the outputs in analog form. It shall be capable of solving these equations in 0.2 sec or less.

7. Memory Loading Operations. The computer shall be capable memory loading from an automatic tape reader, a manual keyboard, or a typewriter. Loading may take place whenever the computer is in the fill mode. Information shall be presented to the computer one character at a time in parallel form. A character shall represent either part of a word to be loaded into the memory or a command used by the computer in the memory lodading process. The computer shall accept data originated by an input device at any rate up to a maximum of 670 characters/sec.

8. Keyboard and Typewriter Information Signals. Information and timing signals shall be available for leading the memory from a keyboard or typewriter.

9. Parity Check. The computer shall check each character received from a tape reader or keyboard/typewriter for parity. The operation shall be performed during both fill and transmit-in modes. A parity error shall inhibit the character from being shifted into the accumulator register, and shall transfer the computer to the parity error state and send a halt signal to the tape reader.

10. Parity Error Signal. A parity error signal shall be generated in the event of a parity error.

11. Verification. While in the fill mode, the computer shall be capable of comparing a word read into the computer from a tape reader or keyboard-typewriter with a selected word in the memory.

12. Verify Error Signal. A verify error signal shall be generated whenever there is a disagreement between the information

following the verify command and the contents of a word in the selected memory location. The signal shall transfer the computer to a verify error state and shall halt the tape reader.

13. Internal Storage. The computer shall have a minimum memory capacity of 4096 words. This will permit solution of the equations in Appendix I, performance of self-diagnostic tests, and provide enough spare capacity for growth during experimental phases.

14. Volatility. The memory shall be non-volatile. All information stored in the memory shall be retained in the event of a power failure or a program protection shutdown.

15. Turn-On and Program Protection Mode. The computer shall be capable of allowing power and environmental anomalies to occur anywhere in a computation, and when corrected, the computer will resume computation without error except for errors due to missing input information.

16. Digital Inputs (from encoders). The computer shall accept digital information in the natural binary coded form or in the two's complement form from at least 7 encoders. Information shall be supplied to the input in parallel form and may contain as many as 22 bits per encoder word.

17. Shaft Encoder Characteristics. Shaft encoders supplying information to the digital input shall employ commutator segments with a natural binary code. Readout of the code shall be by the "V" brush reading method. The least significant bit of the code shall have one output from the lagging brush. Successive bits shall have outputs from a leading brush and a lagging brush. Both leading and lagging pickups shall provide "unprimed" bits. Isolation diodes shall be included in each output line and shall be oriented to pass positive-going signals.

18. Analog Voltage Inputs. The computer shall make available a minimum of 13 input lines for analog voltages from transducers. The voltages shall be unbalanced with respect to ground and shall be between +10.0 v and -10.0 v. The conversion accuracy of an analog voltage to a binary number shall be 0.5 percent of full scale (20 v).

19. Analog Voltage Outputs. The computer shall make available a minimum of 15 analog voltage output lines, and shall be capable of generating and holding an output voltage for each line. The output shall be unbalanced with respect to ground and shall be between +10.0 v and -10.0 v. The conversion accuracy of a binary number to an analog voltage shall be 0.5 percent of full scale (20 v).

20. Intercommunication. The computer shall provide an intercommunication buffer. The intercommunication buffer shall provide the reciprocal transfer of information when directed by an external computer. The intercommunication buffer shall receive the information transfer from another computer.

5. Environmental Requirements

The environmental requirements shall be as follows:

1. Cooling Air Requirements. The computer shall be cooled by means of forced air from an external source. Cooling air for the computer shall be supplied by the using activity. The cooling air will be within a range of temperature between 40 and 80 F. The relative humidity of the cooling air at the input ports will not exceed 50 percent. Cooling air pressure drop shall not exceed 4 in. of water.

2. Pressure - Operating. The computer shall operate without malfunction in an ambient pressure environment of 14.74 ± 3 psi. The computer shall withstand pressures between 10 and 30 psi, changing at a maximum rate of 3 psi per second without permanent degradation. Non-operating: Same as operating.

3. Temperature - Operating. +40 to +130 F ambinet air with cooling air supplied to the computer as indicated. Non-operating: -20 to +160 F ambient.

4. Humidity - Operating. The computer shall suffer no damage, failure or malfunction when subjected to Procedure I of MIL-E-5272C with the following modifications: Temperature +40 to +130 F, relative humidity 80 percent, 5 cycles (120 hr). The computer shall be supplied its normal cooling air during testing. Non-operating: Same as operating, except that the inlet and outlet ports shall be covered during the soak, and the computer may be dried with forced air for 12 hr following the 5th cycle before power is applied.

5. Acceleration - Operating. 2 g in any direction. Non-operating: Same as operating.

6. Vibration - Operating. 2 g, 5 to 40 cps in accordance with MIL-STD-167. Non-operating: 3 g, 10 to 200 cps in accordance with MIL-STD-167.

7. Shock. The computer shall not suffer permanent damage or performance degradation after being subjected to a shock of 10 g in any direction. Non-operating: Same as operating.

6. Control Panel Requirements

A control panel shall be provided which:

1. Initiates power to the computer

2. Controls computer filling from a paper tape reader

3. Provides a keyboard for manual filling

4. Provides manual control over computer modes

5. Indicates computer states

6. Indicates computer halts from high and low voltages; high temperature, and errors in tape reading

7. Provides manual and programed readout of words in either binary coded decimal or actual format

8. Operates from primary power as specified.

7. Software

The following software shall be provided with delivery of the hardware.

a. Test Routines

The test routines are:

1. Go, No-Go Test. A routine shall be provided go, no-go test of the computerl